INTERFACE REQUIREMENTS DOCUMENT (IRD)

for

NATIONAL POLAR-ORBITING OPERATIONAL ENVIRONMENTAL SATELLITE SYSTEM (NPOESS) SPACECRAFT AND SENSORS

Prepared by

Associate Directorate for Acquisition
NPOESS Integrated Program Office

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Approved by	Approved by
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CONTENTS

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Figures

Tables

1.0 SCOPE

1.1 Introduction

The purpose of this Interface Requirements Document (IRD) is twofold. The first is to establish a baseline for interface requirements between the National Polar-orbiting Operational Environmental Satellite System (NPOESS) spacecraft and sensors. Second, it serves as a core building block on which the sensor-to-spacecraft interface can be designed.

The spacecraft-to-sensor interface requirements are broken down into four primary groups: mechanical, power, data, and thermal. A notional diagram of the top-level functional interfaces for any sensor is shown in Figure 1. In addition environmental, software, testing, contamination, launch environment, and safety requirements are defined.

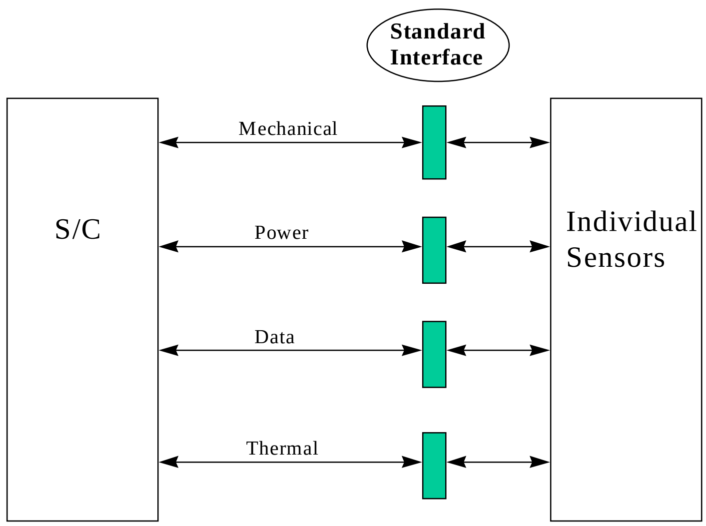


Figure 1. Notional Spacecraft-To-Sensor Functional Interfaces.

This document is intended to provide the basic interface requirements between the sensors, which will be developed first and a spacecraft that will be designed later in the program.

The Spacecraft Contractor and the Sensor Contractors shall each meet their respective interface requirements as defined in this document. Sensor Requirement Documents (SRD) establish the allocation of the system requirements to different NPOESS sensors, and define the sensors' requirements, as well as sensor unique interface requirements. Sensor development contractors shall define their sensor's detailed interface design in an Interface Design Description (IDD) document. The IDD will be used by potential spacecraft integration contractors to evaluate interface and accommodation for their proposed spacecraft. After award of the spacecraft integration contract, each sensor developer and the spacecraft contractor will jointly write an Interface Control Document (ICD) which defines the details of the sensor-to-spacecraft interface and sensor accommodation information.

1.2 System Overview

The U.S. Government currently operates and maintains two polar-orbiting meteorological satellite programs. The U.S. Air Force (USAF) operates the military's Defense Meteorological Satellite Program (DMSP), while the National Oceanic and Atmospheric Administration (NOAA) operates the Polar-orbiting Operational Environmental Satellite (POES) program.

The DoD predecessor program to NPOESS was the DMSP Block 6. Upon completion of Concept Studies started in 1988, two Risk Reduction contracts were awarded in July 1991 to define a military next-generation satellite system (including the Space, Command, Control, and Communications (C^3), and User Segments) to provide meteorological, oceanographic, and solar-environmental support to all DoD users. The purpose of the Risk Reduction effort was to develop preliminary system designs and perform key demonstrations for the baseline system.

The comparable Department of Commerce (DoC) program was the POES Follow-On program, also known as the O, P, Q acquisition. Phase A (advanced study phase) for these satellites was initiated in 1991. Some of the initial design characteristics were common interfaces with the European meteorological operational (METOP) program, growth room to accommodate selected, proven Earth Observation System (EOS) instruments, and a three year design life. The O,P,Q plan was subsequently changed, and the decison was made to procure two additional spacecraft, called N and N-prime based on the TIROS K,L, M spacecraft design. A contract for N and N-prime spacecraft was awarded in December 1994.

In February 1993, the Committee for Science, Space and Technology requested DoD and NOAA to begin looking at opportunities to integrate the DMSP and POES programs and investigate the use of technologies developed by the EOS program. A tri-agency study with DoD, NOAA, and NASA was initiated in June 1993 at the request of Congress. Convergence was also an initiative of the National Performance Review. The result of this triagency study was an agreement to develop a converged operational polar-orbiting environmental satellite system with a transition period beginning in the late 1990s leading to a fully converged system by the mid 2000s. This agreement was formalized by the Office of Science and Technology Policy (OSTP) with the Implementation Plan for a Converged Polar-orbiting Environmental Satellite System, dated May 2, 1994. On May 5, 1994, a Presidential Decision Directive (PDD/NSTC-2) was signed, directing DoD and DoC to converge their independent operational polar-orbiting environmental satellite systems into a single, integrated system. This decision, as part of a National Performance Review recommendation, was expected to save the U.S. Government up to an estimated $300 million in FY94-FY99 with additional savings expected after FY99. A triagency Memorandum of Agreement (MOA), dated May 26, 1995 specifies the roles, responsibilities and agreements between the agencies.

The NPOESS Program is required to provide, for a period of at least 10 years after Initial Operational Capability (IOC), a remote sensing capability to acquire, receive at ground terminals, and disseminate to processing centers, global and regional environmental imagery and specialized meteorological, climatic, terrestrial, oceanographic, solar-geophysical, and other data in support of DOC/NOAA mission requirements, and DoD peacetime and wartime missions.

It is anticipated that operational data will be collected by satellites flying in sun-synchronous near-polar orbits at approximately 833 km altitude with the following nominal nodal crossing times - 0530, 0930, and 1330. Satellites in the 0530 and 1330 orbits are considered U.S. assets and will be developed, acquired, deployed, and operated by the U.S. Satellites in the 0930 orbit are European assets and will be developed, acquired, deployed, and operated by the Europeans. Pending an international agreement, sensors will be exchanged between the United States Government (USG) and the European Organisation for the Exploitation of Meteorological Satellites (EUMETSAT). Under this arrangement, called the Joint Polar System (JPS), some USG sensors will fly on EUMETSAT satellites (designated METOP). In this way, the USG and EUMETSAT requirements will be met jointly by NPOESS satellites and METOP satellites beginning with METOP-3.

It is anticipated that the NPOESS spacecraft will be launched using a medium launch vehicle (MLV or EELV) class of booster. The NPOESS program is comprised of four segments: 1) Space; 2) Launch Support; 3) C^3; and, 4) Interface Data Processor (IDP). Standardization (which includes compatibility, interoperability, interchangeability, and commonality) of DoD, DOC, and NASA systems, components, and interfaces is a primary goal of NPOESS.

1.3 Document Overview

This document comprises five sections.

a. Scope
b. Applicable Documents
c. Requirements
d. Testing Provisions
e. Notes

2.0 APPLICABLE DOCUMENTS

The following documents of the exact issue shown form a part of this IRD to the extent specified herein. In the event of conflict between the documents referenced and the contents of this IRD, the latter shall be the superceding requirement.

2.1 Compliance Documents

MIL-STD-461D	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference
MIL-STD-462D	Measurements of Electromagnetic Interference Characteristics
MIL-STD-882	System Safety Program Requirements
MIL-STD-1540C	Test Requirements for Space Vehicles (Tailored)
MIL-STD-1541A 30 Dec 87	Electromagnetic Compatibility Requirements for Space Systems
MIL-STD-1547A	Electronic Parts, Materials, and Processes for Spacecraft and Launch Vehicles
MIL-STD-1553 21 Sep 78 Notice 1, 12 Feb 80	An Aircraft Internal Time Division Command/Response Multiplex Data Bus
MIL-STD-1773 20 May 88 Notice 1: 2 Oct 89	Fiber Optics Mechanization of an Aircraft Internal Time Division Command/Response Multiplex Data Bus
MIL-A-83577B	Assemblies, Moving Mechanical, for Space and Launch Vehicles
CCSDS 301.0-B-2 April 1990	Consultative Committee for Space Data Systems (CCSDS) Recommendations for Time Code Formats, Blue Book
CCSDS 102.0-B-2 January 1987	Consultative Committee for Space Data Systems (CCSDS) Recommendations for Space Data System Standards, Packet Telemetry, Blue Book
CCSDS 701.0-B-2 October 1989	Consultative Committee for Space Data Systems (CCSDS) Recommendations for Advanced Orbiting Systems (AOS), Networks and Data Links: Architectural Specification
EWR 127-1 31 Mar 95	Range Safety Requirements, Eastern and Western Range

2.2 Reference Documents

ICD GPS 060* 2 Jun 86	GPS User Equipment, Precise Time and Time Interval (PTTI) Interface

MIL-STD-498 5 Dec 94	Software Development and Documentation
MIL-STD-1246C 11 Apr 94	Product Cleanliness Levels and Contamination Control Program
MIL-STD-1522* 18 May 84 Notice 2: 20 Nov 86	Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems
MIL-STD-1543B	Reliability Program Requirements for Space and Missile Systems
MIL-STD-1546B	Parts, Materials and Processes Control Program for Space and Launch Vehicles
MIL-C-24308* 26 Jan 89 Amendment: Jun 93 Supplement 1: 5/93	General Specification for Connectors, Electric, Rectangular, Non-Environmental, Miniature, Polarized Shell, Rack and Panel
MIL-STD-1629*	Procedures for Performing a Failure Mode, Effects and Criticality Analysis
DOD-E-83578A	Explosive Ordnance for Space Vehicles, General Specifications for
DOD-W-83575	General Specification for Wiring Harness, Space Vehicle, Design and Testing
ASTM E595-93 1993	Standard Test Method for Total Mass Loss and Collected Volatile Condensable Materials for Outgassing in a Vacuum Environment, American Society for Testing and Materials
MAN91-2010002	Explosive Safety Standards
40 CFR Parts 1500-1508	National Environmental Policy Act Specifications

3.0 REQUIREMENTS

3.1 Mechanical Requirements

All requirements specified in Section 3.1 shall be met at the mechanical interface and shall be consistent with the SRD allocations.

All interfaces shall be specified in the international system of units, System Internationale (SI), unless design heritage precludes this. Dimensioning shall be in the as-designed units and identified when other than SI.

3.1.1 Sensor Envelopes

3.1.1.1 Sensor Launch Mode Envelope

Sensor components in the launch configuration shall be contained within the sensor launch mode envelope as allocated within the SRD.

3.1.1.2 Sensor On-Orbit Envelope

Sensor components in the on-orbit configuration shall be contained within the sensor on-orbit envelope as allocated in the SRD.

For a sensor with mechanisms that cause a change in the external envelope or external surfaces of the sensor, the initial and final configurations, as well as the swept volumes, shall be documented in the SRD.

3.1.1.3 Sensor Envelope Documentation

The sensor component envelope (including thermal blankets) shall be documented in the ICD by engineering drawings with a set of “not to exceed” dimensions.

3.1.1.4 Stowed and Critical Clearances

The integrating contractor is responsible for defining available sensor volume and making sure the space vehicle fits within the dynamic envelope of the launch vehicle’s fairing. This is controlled with the satellite-to-launch vehicle Interface Control Document (ICD). Both the integrating contractor and the sensor developer must work together to insure that the stowed, deploying, and final deployed positions of the sensor shall clear all obstacles including obstacles on the spacecraft, other sensors, and the launch vehicle. If the sensor is to be deployed, all obstacles shall be cleared in the stowed, deploying, and final deployed positions. If the sensor has moving assemblies, all obstacles shall be cleared within the region of motion. As a baseline, a 2.5 cm clearance between the sensor and surrounding structure shall be maintained. A critical clearance analysis shall be implemented to identify areas where the one inch clearance rule may be violated, accounting for miscellaneous support hardware such as wire bundles and thermal blankets, deflections due to launch loads, launch vibrations, 1-g sag, thermal distortions, and misalignments, with all identified areas tracked in a critical clearance document.

3.1.2 Fields of View

All sensor fields of view shall be within the SRD allocation.

3.1.3 Mass Properties

3.1.3.1 Sensor Mass Allocation

The worst-case sensor mass predicted for the delivered sensor hardware shall be less than or equal to the maximum value allocated for that sensor in the applicable SRD.

3.1.3.2 Sensor Mass Documentation

The mass of the sensor shall be documented in the ICD and shall be measured to +/- 0.1 kg.

3.1.3.3 Sensor Mass Variability

3.1.3.3.1 Sensor Mass Variability Documentation

Sensor mass expulsion rates and substances, if any, shall be documented in the ICD.

3.1.3.3.2 Center of Mass Allocation

Sensors shall be designed to place the center of gravity location as near to the interface plane as possible unless excessive uncompensated momentum precludes this (sometimes the c.g. should be as close to a gimbal axis as possible to reduce uncompensated momentum). The location of the sensor center of mass shall be provided using coordinates based on the space vehicle axes.

3.1.3.3.3 Center of Mass Measurement and Documentation

The stowed and deployed center of mass of each sensor component shall be measured and reported to +/- 5 mm, referenced to the sensor coordinate axes as documented in the ICD.

3.1.3.4 Moments of Inertia

3.1.3.4.1 Moments of Inertia Measurement

The sensor moments of inertia shall be defined using the space vehicle axis convention passing through the sensor center of mass.

3.1.3.4.2 Moments of Inertia Accuracy

Moments of inertia values shall be accurate to within +/-10% (*TBR*).

3.1.3.4.3 Moments of Inertia Documentation

The moments of inertia of each separately mounted component of the sensor shall be documented in the ICD, referenced to the sensor coordinate axes.

3.1.3.4.4 Moments of Inertia Variation Documentation

If the sensor contains movable masses, expendable masses, or deployables, the inertia values for each configuration shall be documented in the ICD.

3.1.4 Mounting

3.1.4.1 Mounting Method

The mounting method shall accommodate manufacturing tolerance, structural, and thermal distortions. The method by which each sensor component is mounted to the spacecraft shall be defined in the ICD.

3.1.4.2 Mounting Interface

3.1.4.2.1 Mounting Interface Documentation

The spacecraft mounting interface for each sensor component shall be documented in the ICD.

3.1.4.2.2 Mounting Hole Coordinates and Dimensions

Coordinates and dimensions of the holes for mounting hardware shall be specified at the mechanical interface and defined in the ICD.

3.1.4.3 Mounting Hardware

3.1.4.3.1 Mounting Hardware Provider

The integrating contractor shall provide all sensor mounting hardware including secondary structures.

3.1.4.3.2 Mounting Hardware Documentation

Sensor mounting hardware shall be defined and documented in the ICD.

3.1.4.3.3 Mounting Surface Requirements

Finish and flatness requirements for the mounting surfaces shall be specified by the integrating contractor and documented in the ICD.

3.1.4.4 Mounting Location and Documentation

The integrating contractor working with the sensor contractor shall determine the location of the sensor on the spacecraft. This location shall be documented in the ICD.

3.1.4.5 Drill Templates

3.1.4.5.1 Drill Template Usage

If drill templates are used for simple planar interfaces, then sensor equipment, spacecraft, and test fixture interfaces shall be drilled using templates.

3.1.4.5.2 Drill Template Fabrication Requirements

The drill template fabrication and functional requirements (e.g. material, use of inserts, etc) shall be provided by the sensor contractor.

3.1.4.5.3 Drill Template Provider

The sensor provider shall provide the drill template to the integrating contractor. The drill template shall include appropriate alignment and location reference information.

3.1.5 Alignment

3.1.5.1 Alignment References

The sensor contractor shall provide a sensor alignment reference. The spacecraft shall provide a spacecraft alignment reference. Both the sensor alignment reference and the spacecraft alignment reference shall be viewable from two orthogonal directions.

3.1.5.2 Alignment Responsibilities

The sensor contractors shall be responsible for measuring the alignment angles between the sensor boresight (line-of-sight), if applicable, and the sensor alignment reference. The integrating contractor shall be responsible for aligning the sensor alignment reference to the spacecraft attitude reference.

3.1.5.3 Alignment Control

The spacecraft contractor shall control the alignment of the sensor alignment reference with respect to the spacecraft attitude reference to within values specified by the sensor contractor.

3.1.5.4 Alignment Knowledge

3.1.5.4.1 Measurement Uncertainty

The spacecraft contractor shall measure the alignment between the sensor alignment reference and the spacecraft attitude reference. The rms uncertainty in the alignment knowledge shall be less than 25 arcsec per axis. This uncertainty shall include (if applicable), but not be limited to, measurement uncertainties, alignment shifts due to vibration environments in both ground processing and launch, uncompensated gravity effects, hygroscopic effects of composite materials, and component removal and replacement.

3.1.5.4.2 Structural Thermal Distortion Uncertainty

The spacecraft contractor shall limit the rms uncertainty in the alignment between the sensor alignment reference and spacecraft attitude reference caused by structural thermal distortion due to the on-orbit thermal environment to be less than 10 arcsec per axis.

3.1.5.5 Spacecraft Attitude Reference

For spacecraft pointing an attitude reference frame shall be defined in accordance with Section 3.1.5.1.

3.1.5.5.1 Attitude Reference Knowledge

The spacecraft will supply a three-axis attitude of the spacecraft attitude reference for ground processing. The supplied attitude will be time-tagged and possess an angular rms accuracy per axis of 10 arcsec over a bandwidth of DC to 10 Hz.

3.1.5.5.2 High Frequency Attitude Reference Errors

The rms of all components of the attitude error of the spacecraft attitude reference with a frequency greater than 10 Hz will be less than 5 arcsec per axis.

3.1.5.5.3 Attitude Reference Control

The rms of the attitude reference control error over a bandwidth of DC to 10 Hz shall be less than 0.01 deg per axis.

3.1.5.5.4 Attitude Reference Rate Error

The rate error of the attitude reference frame shall be less than 0.03 deg/sec during all mission data collection periods.

3.1.5.6 Ephemeris Knowledge

The spacecraft will provide a spacecraft ephemeris estimate with an rms uncertainty of 25/25/25 meters for radial/in-track/cross-track components.

3.1.6 General Structural Design Requirements

A-basis material allowables shall be used for design. An A-basis allowable is defined as a value where 99 percent of a population of values is expected to equal or exceed the allowable, with a confidence of 95 percent.

3.1.6.1 Structural Support

The spacecraft shall provide structural support for the sensor such that the loads transmitted across the interface into the sensor do not exceed interface limit loads to be determined by the spacecraft contractor. The sensor and interface equipment shall be designed to design load factors determined by launch vehicle acceleration levels. A survey of typical launch vehicle environments (accelerations, frequencies, temperatures, etc.) is included in Section 3.9

3.1.6.2 Sensor Structural Dynamics

When the sensor is in its launch-locked configuration, the fundamental natural frequency of the sensor shall be 50 Hz or greater, axial and lateral. For a deployable, the spacecraft integrating contractor shall specify a deployed frequency such that the sensor will not saturate the satellite's control capability. The lowest natural frequency for a deployed sensor shall be greater than 6 Hz (*TBR*). The sensor contractor shall ensure that the sensor dynamic characteristics and control capability (e.g. a gimbaled sensor) will meet the requirements specified for the deployed frequency.

3.1.6.3 Interface Design Limit Loads Requirements

The flight hardware shall be capable of withstanding all worst-case load conditions to which it may be exposed during ground (handling and transportation), pre-launch, launch, and on-orbit operations. Positive structural margins of safety must be maintained so that the sensor can meet all of its design requirements after being subjected to the worst case loads combination. In those cases involving maintenance of sensor critical components for on-orbit operations, the precision elastic limit shall be used for structural materials. The following design factors of safety shall be applied to all loading conditions:

Table 1. Factors of Safety

Design/Test Options	Factors of Safety (Yield)	Factors of Safety (Ultimate)
1. Dedicated test article	1.10	1.25
2. Test on flight article	1.25	1.40
3. Proof test each flight article	1.10	1.25
4. No-static test	1.60	2.00

Note: The level of required analysis increases significantly with increased option number. For the no-static-test option, a detailed and comprehensive structural analysis is required and must be available for review by the space vehicle customer.

The dedicated test article is a qualification test article that will be subjected to the maximum expected loads times 1.25. The test on flight article option refers to a protoqualification on the structure. All composite structures and structural bonded joints shall be proof tested regardless of safety factor, but a metallic structure is usually qualified such that each unit will not have to be tested, or it is protoqualed. The no-static test option allows the capability of the structure to be determined via purely analytical methods, with the analytical models not being verified by test, but verified by the integrator/government for accuracy.

3.1.6.4 Combined Structural Dynamics Analysis

3.1.6.4.1 Combined Structural Dynamics Analysis Responsibility

The integrating contractor shall be responsible for the combined structural dynamics analysis of the spacecraft bus and the sensors.

3.1.6.4.2 Combined Structural/Dynamic Analysis

All models shall be exchanged in NASA Structural Analysis (NASTRAN) bulk data format. A test-verified model is preferred when available, and is required if the sensor lowest frequency is less than 50 Hz as shown by analysis.

3.1.6.4.3 Combined Structural Dynamics Analysis Results

The integrating contractor shall provide the combined structural dynamics analysis results to both the customer program office and the sensor contractors.

3.1.6.4.4 Coupled Loads Analysis Results

The launch vehicle/spacecraft coupled loads analysis will be performed by the launch vehicle contractor. The integrating contractor shall be responsible for providing the results of the launch vehicle/spacecraft coupled loads analysis in a standard format (*TBR*) to the sensor contractors.

3.1.6.4.5 Structural Analyses

A structural analysis using maximum equivalent loads shall be conducted by the sensor developer on all sensors**.** In addition, those sensors with modes under 50 Hz (as shown by the model) must have a full modal survey test completed in a base fixed configuration to obtain all mode shapes and frequencies to correlate the dynamics model.

An analysis using static loads shall be performed if those loads exceed the maximum equivalent values. The integrating contractor shall provide mission-specific information for maximum equivalent loads to the sensor developer for his static load analyses.

3.1.6.5 Pressurized System Design

Sensors with pressurized systems shall follow the requirements of MIL-STD-1522.

3.1.7 Sensor Mass Model

TBR. The requirement for mass models will be determined prior to PDR.

3.1.8 Mechanisms and Deployables

Sensor developers shall use the design and test guidelines provided in MIL-A-83577, to increase reliability of Moving Mechanical Assembly (MMA's) and facilitate integration and test activities.

3.1.8.1 Actuating Devices

Non-explosive actuators shall be preferred over pyrotechnic devices wherever practicable in order to minimize shock loads. A fast release requirement can preclude this design option (paraffin actuators are too slow to release). Actuating circuitry shall be two-fault tolerant to unanticipated deployment or release.

3.1.8.2 Sensor Disturbance Allocations

The sensor developer shall ensure that the "swept" or deployed volume is verified to ease integration and operation, accounting for all distortions and misalignments. In addition, the spacecraft contractor shall provide estimates of allowable disturbance torques, vibration, and end-of-travel or latch-up loads to the sensor developer.

3.1.8.3 Sensor Mechanisms

All sensor mechanisms which require restraint during launch shall be caged during launch without requiring power to maintain the caged condition.

3.1.8.4 Uncompensated Momentum

Each sensor having movable components shall not exceed an uncompensated momentum contribution to be defined and agreed to in an ICD between the sensor contractor and the integrating contractor.

3.1.9 Sensor Disturbance Allocations

3.1.9.1 Constant and Periodic Disturbance Torque Limits

The magnitude spectrum of the disturbance torque that the sensor imparts to the spacecraft shall be in the acceptable region of Figure 2.

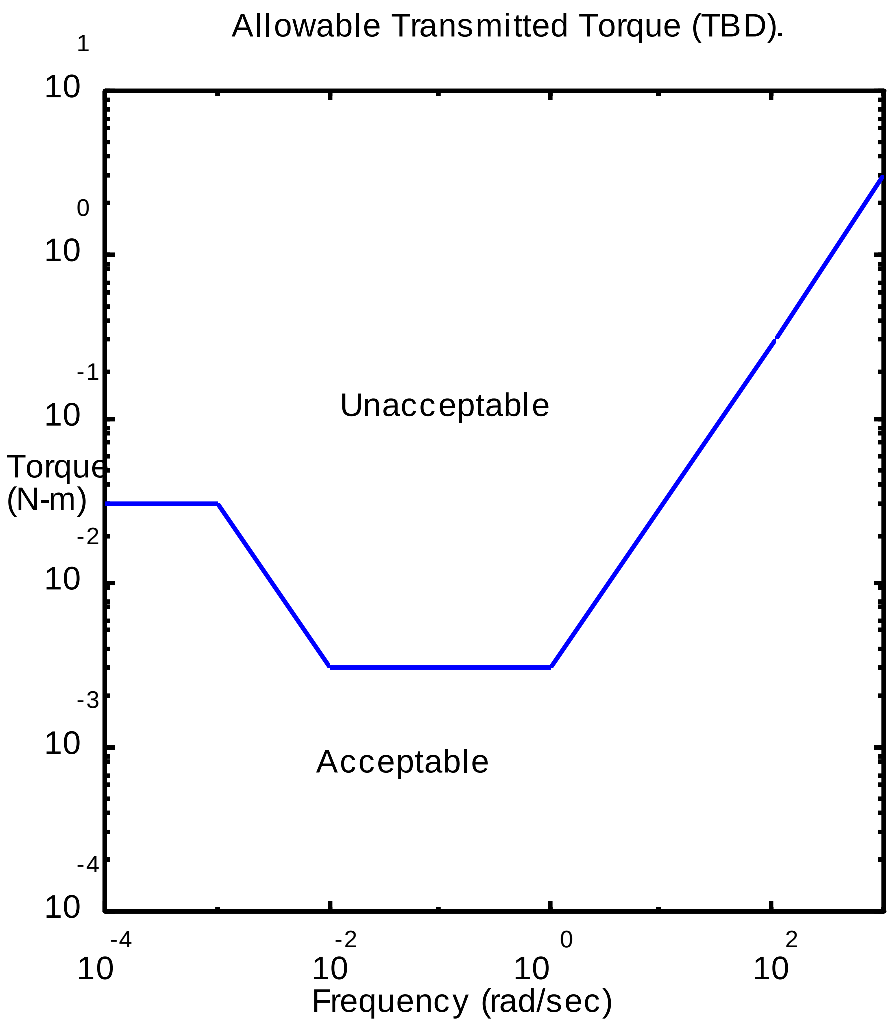


Figure 2 Allowable Transmitted Torque

3.1.9.2 Torque Profile Documentation

The actual sensor torque versus time profile shall be documented in the ICD.

3.1.9.3 Thrust Direction Definition

The magnitude and direction of net thrust resulting from the expulsion of expendables by the sensor shall be documented in the ICD.

3.1.10 Magnetics

Avoid using large quantities of magnetic materials where possible. If magnets are inherent to the sensor design, early estimates of magnetic fields and residual magnetic dipole moments shall be provided to the integrating contractor. A full magnetic survey shall be conducted by the corresponding sensor contractor if the sensor has a total residual uncompensated magnetic moment greater than *TBD* ampere-turn-meter-square.

3.1.11 Access

3.1.11.1 Access Identification

Access requirements shall be documented in the ICD.

3.1.11.2 General Access

All items to be installed, removed, or replaced at the satellite level shall be accessible without disassembly of the unit.

3.1.12 Handling Fixtures

The sensor contractor shall provide proof tested handling fixtures for each component. Handling fixtures shall be designed to 5 times limit load for ultimate and 3 times limit load for yield. Handling fixtures shall be tested to 2 times working load.

3.1.13 Mounting Orientation

Sensors shall be capable of being mounted to the spacecraft with the spacecraft in the horizontal or vertical position.

3.1.14 Sensor to Spacecraft Integration and Test Mounting

Sensors shall be capable of being mounted or removed without removal of other sensors or components.

3.1.15 Non-Flight Equipment

All non-flight items to be installed and/or removed prior to flight shall be identified in the ICD.

3.2 Thermal Requirements

3.2.1 Sensor Thermal Design

All interface requirements specified in Section 3.2 shall be met at the mechanical interface. The sensor thermal design shall provide for:

a. Maintaining the sensor within operating and survival temperature limits,
b. Maintaining the sensor at the minimum turn-on temperature via survival power,
c. Minimizing thermal gradients within the sensor,
d. Thermal decoupling of the sensor from the spacecraft.

3.2.2 Thermal Isolation to Spacecraft

The spacecraft shall not be used primarily as a heat source or sink (i.e., the sensor design should maximize thermal isolation). Sensor components shall be designed to maintain the sensor within its allowable temperature limits. The thermal control units shall be mounted on the sensor, where possible, or insulated in order to minimize thermal load to the spacecraft.

3.2.3 Heat Transfer

3.2.3.1 Heat Transfer to Spacecraft

The heat transfer between the sensor and the spacecraft shall not exceed 10.0 watts (*TBR*) maximum. Sensors with high power dissipation near the interface, or configuration requirements that do not lend themselves to thermal-isolation methods, require the contractor to develop mission sensor-specific heat-transfer rates. For design purposes, the 10.0 watts heat transfer shall be applied in a worst-case scenario.

3.2.3.2 Radiation

Incident radiation between the spacecraft and a sensor on any given surface shall be minimized. The spacecraft contractor shall provide the radiative loads to the sensor. The environmental fluxes, as shown in Table 2 below, shall add solar, albedo and earth IR hot fluxes for the hot case analysis and cold fluxes for the cold case analysis.

Table 2 Worse-Case Hot and Cold Environments

	Hot Case		**Cold Case**	
	BTU/hr-ft^2	**W/m^2**	**BTU/hr-ft^2**	**W/m^2**
Solar Radiation	444	1400	415	1308
Albedo	172	542	86	271
Earth IR Radiation	83	262	60	189

3.2.4 Temperature Ranges

3.2.4.1 Spacecraft Temperature Range

For planning and preliminary design purposes, the interface temperature of the spacecraft shall be initially assumed to range from:

a. 5 °C (*TBR*) to +40 °C during normal operations
b. -20 °C to +50 °C during survival modes

3.2.4.2 Sensor Temperature Range

Temperature limits for sensor components during ground test and orbital operations shall be documented in the ICD. Operating, non-operating, survival and turn-on temperature requirements shall be included.

3.2.4.3 Thermal Uncertainty Margins

Thermal uncertainty margins used during the design and validation shall be applied to determine acceptance ranges per MIL-STD-1540C. If heaters are employed, a 25% heater control authority can be used in place of the thermal uncertainty margin. Protoqualification ranges shall be calculated by adding an additional margin of ±5 °C.

3.2.5 Temperature Monitoring

3.2.5.1 Mechanical Mounting Interface Temperature Monitoring

The spacecraft shall monitor and report in the spacecraft telemetry the temperature of the spacecraft at the sensor mechanical mounting interfaces.

3.2.5.2 Sensor Temperature Monitoring

All critical sensor temperatures shall be measured and reported in the health and status telemetry data.

3.2.5.3 Temperature Sensor Locations

The location of all sensor and mounting interface temperature sensors shall be documented in the ICD.

3.2.6 Thermal Control Design

3.2.6.1 Survival Heater Design

Sensors shall use survival heaters to maintain temperature at the safe turn-on level. Operational heaters shall be controlled by the sensor.

Electrical power for survival heaters shall be provided by the spacecraft and accommodate at least <u>two strings,</u> a primary and secondary string, of sensor survival heaters. Survival heater circuits shall not exceed 0.5 amperes per string, and shall be provided directly to thermostatically controlled heaters on the sensor side. These heaters must be capable of operation when the sensor power is off. The interface shall also have the capability to accommodate up to five analog thermistors (3 kilo-ohm at 25 °C) per interface. These analog lines are separate and in addition to any state of health (SOH) input being transmitted over the serial data bus interface and are intended to provide insight during periods when the sensor power is off; therefore, excitation of thermistors shall be provided by the spacecraft.

When the sensor is powered, the sensor shall be responsible for distributing power and controlling the operational heaters inside the sensor. When the sensor is unpowered, the survival heaters shall nominally be controlled by the spacecraft through the sensor thermistor inputs. Redundant thermostats shall be used.

3.2.6.2 Thermal Control Hardware

Thermal control hardware shall be documented in the ICD. The responsibility for providing the thermal control hardware is defined in Table 3

Table 3. Thermal Control Hardware Responsibility

Hardware	Responsibility
Survival Heaters	Sensor Provider
Sensor, Thermal Control Hardware, including blankets, louvers, and heat pipes	Sensor Provider
Thermal insulation Blankets to Interface between the Sensor Thermal Blankets and the Spacecraft Thermal Blankets	Spacecraft Provider

3.2.6.3 Multilayer Insulation

Multilayer Insulation (MLI) used in thermal control design shall have the following provisions: venting, interfacing with spacecraft thermal control surfaces, and electrical grounding to prevent Electro Static Discharge (ESD). The integrating contractor shall approve the MLI selection in the PMPCB (Parts, Materials and Processes Control Board) review process.

3.2.6.4 Other Considerations

Thermal control surfaces shall be cleanable to visibly clean or better. Any sealed or closed system such as heat pipes, thermal control enclosures or fluid loops shall be analyzed to demonstrate that a safety hazard does not exist.

3.3 Electrical Power Requirements

3.3.1 Electrical Interfaces

The electrical interfaces (Figure 3) shall include the following:

a. Operational Power Interface
b. Survival Heater Power Bus
c. Pulse Command Interface
d. High-rate Data Bus
e. Command Telemetry and Low-rate Data Bus
f. Grounding Interface
g. Test Point Interface

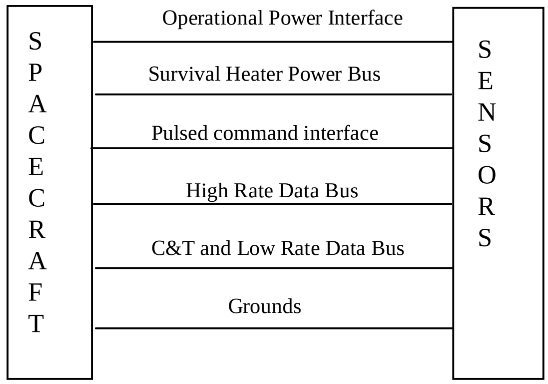


Figure 3. Spacecraft-Sensor Electrical Interfaces

3.3.2 Electrical Voltage

3.3.2.1 Primary Sensor Voltage

The electrical power supplied at the sensor interface shall be regulated 28 $\pm$ 6.0 Vdc (*TBR*).

3.3.2.2 Voltage Ripple

The power source-generated and load-induced ripple, including repetitive spikes, shall not exceed 1.0 volt peak-to-peak as measured over the bandwidth of 30 Hz to 1.0 kHz, and 0.5 volt peak-to-peak from 1.0 kHz to 10 MHz when the power system is delivering the maximum rated current into loads.

3.3.2.3 Reflected Ripple

Loads shall not produce reflected ripple greater than the limits of MIL-STD-461D, part 3, CEO1 and CEO3. CEO1 maximum levels apply to loads that are 15 amps/450 watts and greater. CEO1 maximum emissions shall be reduced by 20 dB for each 20 amps reduction in current.

3.3.2.4 Transients

Positive and negative voltage surges shall decay to within steady state limits in less than 5 and 100 milliseconds, respectively. All spacecraft and sensor components shall remain undamaged when subjected to step changes of the input voltage from 0% to 140% and from 120% to 0% of the nominal load voltage (28 volts). The step changes, exclusive of spikes, are the instantaneous surge amplitudes produced by load switching and the clearing of faults on the space-vehicle power bus. With step changes from 0% to 100% of the nominal load voltage, the instantaneous inrush current shall not exceed 4-times the maximum average input current.

3.3.2.5 Undervoltage Protection

The spacecraft shall be able to remove bus power to all sensors if the bus voltage drop below 22 volts. Control heaters shall also be turned off during these occurrences. This does not apply to survival heaters.

3.3.2.6 Spacecraft Power Bus Impedance

The spacecraft bus impedance at the interface looking back into the source shall be less than 100 milliohms resistive and 5 micro-henries inductive.

3.3.3 Electrical Current

Three types of power shall be supplied to each sensor:

3.3.3.1 Survival Heater Power

Direct bus connection through a 5 ampere spacecraft fuse for 50 watt heater maximum (*TBR*). The sensor must have two series thermostats to ensure fault-tolerant usage of this bus.

3.3.3.2 Control Heater Power

Bus connection is made through a 5 ampere fuse and relay switch in the spacecraft for a 50 watt heater load maximum (*TBR*).

3.3.3.3 Equipment Power

Two supply circuits types shall be provided:

a. A 0-5 ampere steady-state power connection is made through a 15 ampere fuse and relay switch in the spacecraft. Peak inrush at initial power application is 10-times steady-state. Peak inrush at equipment power-on is 4-times steady-state.

b. A 5-20 ampere steady-state power connection is made through a 60 ampere fuse and relay switch in the spacecraft. Peak inrush at initial power application is 4-times steady-state. Peak inrush at equipment power-on is 4-times steady-state.

3.3.4 Grounds, Returns, and References

3.3.4.1 Grounding

The spacecraft and sensors shall isolate space-vehicle primary power from chassis, telemetry, and secondary power by more than 1 Meg ohm. The primary power source (battery, power converter) shall be chassis grounded at only one point to avoid large structure current flow which might interfere with other spacecraft loads.

3.3.4.2 Power Leads and Signal Returns

The method used to reference the signal back to the secondary power return is dependent on the signal type. The goal is to minimize the voltage drop across the return. Secondary power and signal returns shall be isolated from the primary power return by not less than 1 Meg ohm when the sensor is disconnected from the interface/spacecraft and when measured at the sensor input. The secondary grounds may be grounded to structure if the local structure is conductive.

3.3.4.3 Power Harnesses

3.3.4.3.1 ElectroMagnetic Interference/Compatibility (EMI/EMC) Considerations.

Data and telemetry signals shall be segregated and routed from any power circuitry via a separate connector.

3.3.4.3.2 Fault Isolation

Fault isolation shall be included on the spacecraft side of the interface. The spacecraft shall be capable of removing any load in excess of 30 watts. The fault isolation shall either open the circuit to remove the load and short circuit from the spacecraft, or limit the current to the maximum specified load current. Fuses and circuit breakers shall be sized to protect wire between the bus and the sensor. The wire shall be sized to the maximum load. The fuses shall be derated by a factor of three.

3.3.4.3.3 Electrical Connectors

For the standard electrical connector, separate D connectors, such as Cannon NM-K52 (rated at 5 amps, derated below 5 amps), military D subminiature nonmagnetic/no-outgas connectors as described in MIL-C-24308, or Positronic SAD Series connector (rated at 7.5 amp, derated below 7.5 amps for use when load requires 5 amps) and Kem connector accessories shall be used for primary power, survival heater power, and analog thermistor returns. All interface circuits shall be categorized by signal type using DOD-W-83575 as a guide. Primary and redundant connectors shall be differentiated by clearly marking all boxes and cables. Interface requirements for sensor electrical connectors are as follows: (*TBR*). All spacecraft interface mating connectors shall be provided by the spacecraft contractor.

3.3.4.3.4 Wiring

All power harnesses shall be at least #20 AWG, with 150 °C insulation, and twisted pairs to reduce magnetic contribution. The wire current-handling capability shall be calculated at the ambient temperature. The contractor shall determine the proper wire insulation requirements for any wire directly exposed to the space environment.

3.3.4.3.5 Power Cabling

Power cables, supply and return wires shall be twisted to reduce electro-magnetic contribution.

3.3.5.3.6 Signal Cabling

The voltage drop across any secondary return shall be less than the maximum allowable noise on the signal circuit reference. Digital or analog cross talk between any two signal lines in data connectors shall be no greater than -20 dB at the maximum data rate.

3.3.5.3.7 Electromagnetic Interference Filtering of Space Vehicle Power

The sensor shall have EMI input filters installed on the sensor side of the power interface. This does not apply to the survival heater circuits which are controlled on the spacecraft side of the interface. The filters shall provide both common-mode and differential-mode filtering capable of meeting EMC requirements per MIL-STD-1541A. The filters shall be designed to withstand and suppress electrical transients.

3.3.6 Test Points

TBD.

3.3.7 Spacecraft/Sensor Interface Simulator

The sensor contractor shall provide a simulator to the spacecraft contractor for initial interface testing. The simulator shall, as a minimum, have the same mechanical characteristics (geometry, mass, c.g., mounting holes, connectors, etc) as the real sensor-to-spacecraft interfaces. In addition, an electrical interface simulator, having corresponding connectors and pin assignment, shall be provided.

3.4 Command and Data Handling (C&DH) Requirements

3.4.1 Notional Functional and Performance Description

The spacecraft C&DH subsystem shall perform the following functions:

- Collect all mission data, vehicle health and status, and receive and process ground commands and memory loads.
- Format, process, and store the collected data for both real-time and stored data transmission. Transfer mission data to the communications subsystem for real-time and stored data transmission.
- Maintain a data base of allowable limits for the spacecraft and sensors to establish nominal and maximum/minimum values for monitoring status and health.
- Monitor all the vehicle health and status telemetry (such as spacecraft attitude, temperature ranges, and solar array pointing) and issue commands to the subsystems for appropriate action.
- Receives demodulated uplink commands and memory loads from the Communication subsystem, decodes the commands and transmits them to the appropriate destination via the MIL-STD-1553 data bus.
- Initiate power down or self-recover mode if required to maintain spacecraft power and ground communication. In order to perform these functions, the C&DH shall interface with the sensors and subsystems via:
    - Direct hardwire for pulse commands
    - Command, Telemetry and Low Rate Mission Data Bus
    - High Rate Mission Data Bus

3.4.2 Satellite Modes

The satellite shall implement the following common modes as a minimum:

- OFF Mode
- OPERATIONAL Mode
- SAFE HOLD Mode
- AUTONOMOUS Mode
- DIAGNOSTICS Mode

Additional modes such as a LAUNCH, EARLY ORBIT, and/or CAL/VAL Modes may be needed.

3.4.2.1 OFF Mode

In the OFF Mode, no power shall be supplied to the sensor.

3.4.2.2 OPERATIONAL Mode

The sensors shall have one or more OPERATIONAL Modes for collecting data as defined in the applicable Sensor Requirements Document. The sensor shall be fully operational in this mode. Additional modes may be required for autonomous, backup, or contingency situations.

3.4.2.3 SAFE HOLD Mode

In the event of an anomalous spacecraft or sensor situation, it may be necessary to enter the SAFE HOLD Mode to conserve power. A power subsystem anomaly is such an event. The C&DH shall issue power conservation re-configuration commands to the sensors via the 1553 data bus which will place the sensor in a safe configuration. Ground intervention shall be required to return to OPERATIONAL Mode.

3.4.2.4 AUTONOMOUS Mode

In the AUTONOMOUS mode, the satellite shall be capable of operating up to 60 days without additional commands.

3.4.2.5 DIAGNOSTICS Mode

Diagnostic mode shall include housekeeping, troubleshooting, testing, and software updates.

3.4.2.6 Mode Documentation

Additional detail on the various satellite modes shall be defined in the ICD. SAFE HOLD re-configuration commands shall be defined in the ICD.

3.4.3 General Electrical Interface Requirements

Unless specified otherwise, the following requirements apply to all C&DH electrical interfaces.

3.4.3.1 Interface Conductors

All signal interfaces shall use shielded conductors. Conductors may include, but are not limited to, twisted pair, coaxial, twin axial, dual coaxial types, and fiber optics.

3.4.3.2 Interface Circuitry Isolation

The sensor shall maintain electrical isolation of greater than 100 kilo-ohms between the primary and redundant interface circuitry within the sensor front end.

3.4.3.3 Interface Fault Tolerance

The sensor and spacecraft bus shall be tolerant of a single fault occurring in a signal interface circuit on either side of the interface.

3.4.4 Command, Telemetry (C&T) and Low-Rate Data Bus Requirements

3.4.4.1 Bus Functions

The C&T and Low-Rate Data bus shall be used as shown below in Figure 4:

a. Spacecraft to sensor/remote terminal transfers consisting of:
   - real time commands
   - stored commands
   - memory loads
   - frame sync and time code data

b. Sensor/remote terminal to spacecraft transfers consisting of:
   - sensor health and status telemetry
   - sensor diagnostic data
   - low rate science data

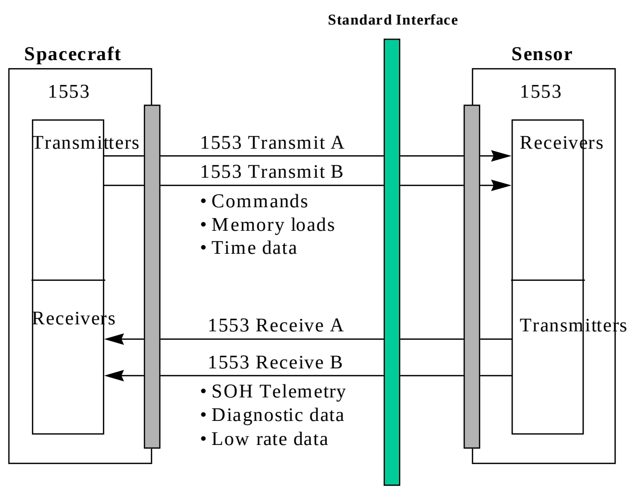


Figure 4. Data Transfer Interface

3.4.4.2 Bus Type

The C&T and Low-Rate Data bus shall be a dual standby redundant data bus that fully complies with the requirements of MIL-STD-1553B, Notice 2, all sections (*TBR*). Additional requirements shall be specified wherever necessary to select MIL-STD-1553 options and to eliminate ambiguities. MIL-STD-1773 is being considered as an option for the C&T and Low-Rate Data bus.

3.4.4.3 Bus Configuration

The spacecraft C&DH shall perform the Bus Controller (BC) function for the 1553 data bus to send data to and collect data from the sensors. Spacecraft subsystems shall interface with the 1553 data bus via a Remote

Terminal (RT) as shown in Figure 5. Those sensors without an internal 1553 interface shall also interface to the data bus via a Remote Terminal. The sensors shall interface to the dual standby redundant data bus via dual redundant RT(s) to receive data from and send data to the spacecraft upon request.

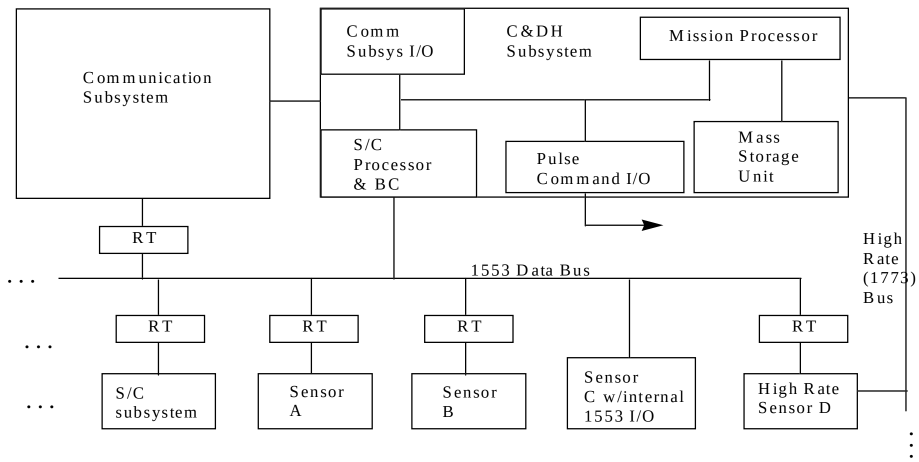


Figure 5. Command and Data Handling Interface Topology

3.4.4.4 General Bus Requirements

3.4.4.4.1 Electrical Interface

Each electrical interface between the sensor/RT and the data bus shall be dual redundant. Each functionally distinct RT shall be dual redundant. Each RT shall be individually transformer coupled to both the primary and the redundant data buses.

No single failure in the data bus electrical interface circuit on either the sensor/RT side of the interface or the spacecraft data bus side of the interface shall cause the sensor to lose the capability to communicate with either the primary or the redundant data buses via each functionally distinct RT.

3.4.4.4.2 Data Bus Monitoring

The Bus Controller shall have the capability to monitor the bus status and switch to a redundant bus so that no sensor/remote terminal or single data bus failure shall prevent the Bus Controller from maintaining data flow over the Data Bus.

3.4.4.5 Sensor Commands and Memory Load

3.4.4.5.1 Command Types

The spacecraft shall deliver the following data to the specified sensor RT-receive subaddresses by conducting single BC to RT Transfers or single RT to RT Transfers (from a spacecraft RT to an sensor RT). The C&DH shall have the capability to issue on/off pulse commands to the sensors via a redundant hardwire interface. The sensor shall be capable of accepting pulse and serial commands with the characteristics specified:

Pulse Command

a. Logic 0	*TBR*
b. Logic 1	*TBR*
c. Load Capacitance	*TBR*
d. Pulse Width	*TBR*
e. Voltage Rise Time	*TBR*
f. Voltage Fall Time	*TBR*
g. Noise Immunity	*TBR*
h. Inductive Spike Suppression	*TBR*

Serial Command

The serial command input shall consist of NRZ data, clock and envelope signals. The RT to Sensor serial command transfer shall consist of a three wire interface. Characteristics of the interface are *TBR*.

3.4.4.5.2 Packetization for Commands and Memory Loads

Unless otherwise specified, all commands and memory loads delivered to the sensor/remote terminal shall be formatted in accordance with the CCSDS Telecommand packet defined in CCSDS 203.0-B-1.

3.4.4.5.3 Documentation

All sensor commands and memory load packet descriptions shall be documented in the ICD.

3.4.4.5.4 Critical Commands

Initiation of critical or hazardous functions shall use, as a minimum, separate enable and execute commands to prevent inadvertent execution of critical commands.

3.4.4.6 Frame Synchronization and Time Code Data

The spacecraft shall provide frame sync and time code data signals to the sensors via the 1553 Data Bus using a BC to RT Broadcast message as described in MIL-STD-1553, Notice 2.

The format of the vehicle time code words shall be based on the GPS UTC time representation. On-board absolute correlation of time shall be 1 millisecond or better with a correlation to 1 microsecond as a goal. Time representation shall be transmitted over the 1553 data bus once per second and shall correspond to the time of the rising edge of the time-of-day pulse.

3.4.4.7 Health and Status Telemetry and Diagnostic Data

3.4.4.7.1 Telemetry Data Overview

The spacecraft computer within the C&DH shall hold in memory the Health and Status Telemetry formats. The capability to select, by command, one of *TBD* fixed formats and dwell mode shall be provided. The spacecraft shall collect the selected format's telemetry data by conducting a sequence of RT to BC transfers. The collected data includes the following types: unconditioned analog, conditioned analog, unconditioned bilevel, conditioned bilevel, and serial digital. Characteristics of these signals are *TBD*.

All critical telemetry channels shall be redundant.

3.4.4.7.2 Health and Status Telemetry Data

Sensor health and status telemetry data shall include housekeeping data required for sensor status and health monitoring at the Ground Control Center. Sensor health and status telemetry includes:

- Sensor mode and configuration
- Sensor temperatures
- Sensor power supply current and voltage
- Relay status, scan mirror rotation and other rotating mechanism rates
- Other telemetry data required to support sensor performance evaluation

3.4.4.7.3 Telemetry Diagnostic Data

During sensor anomaly resolution, the spacecraft C&DH shall have the capability to dwell on particular telemetry measurands within the selected telemetry format in support of ground diagnostic investigation of the sensor anomaly. Dwell capability shall be a ground initiated process.

3.4.4.8 Low Rate Data

Low rate data is defined as the user mission data from the sensors identified to produce output data rates less than 100 kbps. The spacecraft C&DH shall collect the low rate data from the respective sensors through a sequence of data transfers over the 1553 data bus. This data is formatted for downlink by the C&DH and is transferred either to the Communication Subsystem for direct downlink to the users or stored in a mass storage unit within the C&DH for a subsequent transfer to the Communication Subsystem when in view of the appropriate ground site.

3.4.4.8.1 Telemetry and Low Rate Data Packetization

All telemetry and low rate data shall be packetized using the CCSDS Path Protocol Data Unit format as defined in CCSDS 701.0-B-1.

3.4.4.8.2 Data Bus Sampling Rate

The combined rate at which the spacecraft transmits commands, samples telemetry and collects low-rate mission data to/from the sensors/subsystems, the maximum duration of a data transfer cycle and the minimum time gap between transfer cycles shall comply with the MIL-STD-1553, Notice 2 specification. The bus sampling rates for each sensor shall meet the sensor functional requirements as identified in the Sensor Requirements Document.

3.4.4.9 High Rate Data Bus

3.4.4.9.1 Bus Functions

Redundant High-Rate data buses shall be used to transfer data from a high-rate sensors or sensor suites to the spacecraft C&DH subsystem. The C&DH mission computer will store this data in a mass storage unit and transfer it to the Communication Subsystem for downlink.

3.4.4.9.2 High Rate Data Bus Transmission Rate

The high-rate data bus shall be used for a sensor with data rates of >100 Kbps or to a sensor suite with combined data rates of >100 Kbps.

3.4.4.9.3 Bus Type

The High rate data bus shall be in compliance with MIL-STD-1773B (*TBD*).

3.4.4.9.4 High Rate Data Packetization

All data to be transferred to the spacecraft C&DH via the high rate data bus shall be packetized using the CCSDS Path Protocol Data Unit format defined in CCSDS 701.0-B-1.

3.5 Contamination

A contamination control program shall be developed and implemented as part of the sensor to spacecraft interface. The contamination requirements shall be included in the ICD. A system level contamination plan shall be developed by the integrating contractor. Sensor and other sensor contamination and cleanliness requirements shall be considered along with the spacecraft requirements and the resulting contamination budget provided to each sensor developer.

3.5.1 Contamination Control Requirements

The spacecraft and sensor contractors shall perform independent contamination analyses to identify, locate and size components sensitive to contamination and assess, calculate or measure the maximum allowable particulate and molecular film (nonvolatile residue, or NVR) contamination consistent with top level mission performance and lifetime specifications. The contamination limits shall be documented in the ICD

3.5.2 Sensor Sources of Contamination

Sensor contractors shall identify and characterize all sources of contamination that can be emitted from the sensor. At a minimum, the characterization shall include the material name, the amount, the emission rate, and its location. The extent to which outgassing products have access to exterior surfaces shall also be considered. Data from the ASTM E-595 test for percent total mass loss (%TML) and percent collected volatile condensable material (%CVCM) shall be used. The outgassing chemical species and any tendency to photodeposit in a UV or energetic particle radiation environment shall be identified and quantified.

Material outgassing is an issue that must be coordinated between the sensor developer and the satellite contractor. Materials and coatings known to flake or outgas, such as cadmium and zinc, shall be avoided. Materials with the following properties are recommended: Total Mass Loss (TML) less than 1.0%; production of Collected Volatile Condensable Material (CVCM) less than 0.1% when tested under conditions of ASTM E595-93 or equivalent. Composite materials are an exception.

If voltages over 60 (*TBR*) V are present, the design shall be protected (e.g. potting, pressure vessel) and tested for arcing.

3.5.3 Sensor Venting

Sensor contractors shall define the location, size, path and operation time of vents in the sensors. This information shall be defined in the ICD.

3.5.4 Sensor Purge Requirements

Spacecraft and sensor purge requirements, including type of purge gas, flow rate, gas purity specifications, filtration and desiccant requirements, and the acceptable limits for purge interruption, shall be provided and documented in the ICD.

3.5.5 Sensor Inspection and Cleaning During I&T

Inspections and cleaning by the spacecraft and sensor contractors during Integration and Test (I&T) shall be coordinated and defined in the ICD.

3.5.6 Spacecraft Contractor Supplied Analysis Inputs

As part of the contamination control analysis and ICD development, the spacecraft contractor shall provide the plume flowfield analyses for all spacecraft thrusters. The analyses shall include the identity and quantity of each chemical species emitted and provide sufficient dynamic information to determine the final deposition amount on all sensitive surfaces. Margin shall be given for an additional contribution from the payload fairing and launch vehicle thrusters fired after the payload fairing has been jettisoned.

The spacecraft contractor shall also identify and characterize all sources of contamination that can be emitted from the spacecraft. At a minimum, the characterization shall include the material name, the amount, the emission rate, and its location. Data from the ASTM E-595 test for percent total mass loss (%TML) and percent collected volatile condensable material (%CVCM) shall be used. The outgassing chemical species and any tendency to photodeposit in a UV or energetic particle radiation environment shall be identified and quantified.

3.5.7 Atomic Oxygen Contamination

The spacecraft and sensor contractors shall consider the effects of atomic oxygen in the space environment. spacecraft and sensor materials selection should minimize the generation of particulate and molecular film contamination via interaction with atomic oxygen.

3.5.8 Facility Environmental Requirements

The spacecraft and sensor contractors shall describe the required integration and test environments using the definitions of FED-STD-209E or ISO 209. The sensors shall be integrated with the spacecraft in a Class 10,000 cleanroom environment and maintained in that environment as much as possible during the integration and test flow. The facility requirements shall be documented in the ICD and should include air cleanliness, air flow and recirculation rates, temperature and humidity, and tolerance for out-of-spec conditions (i.e., intermittent spikes) as a minimum. Requirements shall include verification by standard testing methods to be performed at regular, specified intervals.

3.5.9 GSE Cleanliness Requirements

Spacecraft and sensor contractors shall document the need for contamination control of all Ground Support Equipment (GSE) entering cleanrooms. In addition, all GSE used inside thermal/vacuum facilities shall be cleaned and verified as vacuum compatible.

3.6 Software and EGSE Requirements

3.6.1 Software Programming Language Requirements

The sensor software provider should implement all software using standard Ada (MIL-STD-1815A), Fortran (ANSI STD X3.9-1978), or C (ANSI STD X3/159-1989) or other high level languages (*TBR*).

3.6.2 Sensor Flight Software Requirements

3.6.2.1 Sensor Flight Software Version Control

All software and firmware shall be implemented with an internal identifier (embedded in the executable program) that can be included in the sensor engineering data. This identifier shall be keyed to the configuration management process so that the exact version of software and firmware residing in the sensor can be determined at any time.

3.6.2.2 Sensor Flight Software Loading

Loading of the sensor microprocessor via hardline shall take no longer than 10 minutes following a hardware reset or power-up.

3.6.2.3 Sensor Flight Software On-Orbit Installation and Verification

Flight software shall be designed so that complete or partial revisions can be installed and verified on-orbit.

3.6.3 Sensor Ground Support Equipment (GSE) Software Requirements

Commands that can potentially damage hardwareor cause injury to personnel shall require test operator authorization prior to being sent to the sensor for execution.

3.6.4 Sensor GSE to Spacecraft I&T GSE Interface

The electrical sensor GSE (IGSE) shall interface with the spacecraft electrical GSE via a local area network (for example, ETHERNET) and/or point-to-point links if necessary for spacecraft level testing. IGSE shall receive sensor telemetry via the spacecraft electrical GSE. Commanding of the sensor shall be from the spacecraft electrical GSE.

3.7 Environmental Requirements

3.7.1 Total Ionizing Dose Environment

The sensor and the spacecraft shall be capable of meeting the proton and electron total dose levels for a 7-year mission are given in the table below. Two times the total dose shall be used to provide a design margin factor of two (Note: aE+N=a x 10^N, e.g. 3.264E+06 = 3.264 x 10^6; one mil is 10^{-3} inch).

SHIELDING Mils (AI)	Trapped Protons/7 Yr	Trapped Electrons/7 Yr
100	6.50 E03	1.81 E04
200	4.79 E03	2.06 E03
400	3.67 E03	6.76 E01
600	3.05 E03	4.35 E01
1000	2.25 E03	3.04 E01

3.7.2 Cosmic Ray and High Energy Proton Environment

3.7.2.1 Single Events Radiation Environment

The sensor and the spacecraft shall be capable of meeting all performance requirements in the Cosmic Ray and High Energy Proton Radiation Environment specified in 3.7.2.1.1 and 3.7.2.1.2. Predictions of single events (i.e. single event latch-up, single event upset and single event burn-out) induced by galactic cosmic ray ions and high energy protons shall be performed separately and the results combined.

3.7.2.1.1 Galactic Cosmic Ray Linear Energy Transfer (LET) Spectrum

The integral galactic cosmic ray linear energy transfer spectrum in *TBS* shall be used for prediction of ion-induced single events.

3.7.2.1.2 High Energy Proton Fluence

The differential proton fluence in *TBS*, which consists of trapped protons and galactic cosmic ray protons shall be used for prediction of proton-induced single events in the absence of solar flares.

The differential proton fluence in *TBS*, which consists of trapped protons, galactic cosmic ray protons and solar flare protons, shall be used for prediction of proton-induced single events with solar flares.

3.7.2.1.3 Peak Fluxes

The sensor and spacecraft shall be capable of meeting all performance requirements when exposed to trapped proton ($E \geq 5$ MeV) flux of *TBS* particles/cm^2 sec, trapped proton ($E \geq 0.5$ MeV) flux of *TBS* particles/cm^2 sec with the following estimated solar flare proton peak fluxes and associated total event integral fluences for each extremely large solar flare:

Energy (MeV) (Particles/cm)	Flux (Particles/cm^2sec)	Total Event Integral Fluence
>10	*TBS*	*TBS*
>30	*TBS*	*TBS*
>60	*TBS*	*TBS*
>100	-	*TBS*

The total event integral fluence is accumulated within a time interval of a few hours to two days.

3.7.2.2 Displacement Damage

Prediction of proton-induced displacement damage (also known as the bulk damage) to Charge Coupled Device (CCD) detectors shall be based on the differential proton fluence in *TBS*.

Where CCD detectors are used, the design shall incorporate features that minimize the effects of displacement damage.

3.7.3 Atomic Oxygen

The sensor shall meet performance requirements during exposure to atomic oxygen (AO) experienced during a 833 km polar orbit for seven years. Atomic oxygen fluence is shown in *TBS*.

3.7.4 Electromagnetic Compatibility

3.7.4.1 General

There are six macro-level interfaces to consider for EMC:

1) Interface between sensors and spacecraft bus
2) Interface between spacecraft and external environment
3) Interface between spacecraft and launch vehicle
4) Interface between spacecraft and ground support equipment.
5) Interface between spacecraft bus/sensors and test equipment.
6) Interface between sensors and launch vehicle

There are four EMC interfaces:

1) Conducted Emissions/Susceptibility
2) Radiated Emissions/Susceptibility
3) Grounding
4) Wiring

3.7.4.2 Baseline Requirements

3.7.4.2.1 System Electromagnetic Compatibility

The spacecraft bus, sensors, ground support equipment, and test equipment shall operate within acceptable limits, i.e., without performance degradation with each other, and the external environment.

3.7.4.2.1.1 Search and Rescue Sensor Compatibility

The Search and Rescue sensor shall not be impacted at any of its receiver frequencies.

3.7.4.2.2 Interface Margins

Each interface margin shall be the larger of the following:

1) At least 12 dB.
2) At least large enough to cover manufacturing variations from spacecraft to spacecraft and end-of-life variations.

Electroexplosive devices circuits shall have at least a 20 dB margin.

3.7.4.2.3 Frequency Management

All intended receivers and transmitters shall have frequency assignment and allocation in accordance with all National Telecommunications and Information Administration regulations.

3.7.4.3 External Environment

3.7.4.3.1 External RF Environment

The system shall operate without performance degradation in the following external environment.

Frequency	V/m
10 k-100M	*TBD*
100 M - 1 G	*TBD*
1 - 10 G	*TBD*
10 G - 40 G	*TBD*

The intended receivers shall operate without performance degradation for the external environment outside their pass band and shall survive and automatically recover for the external environment inside their pass band.

3.7.4.3.2 Lightning

The system shall be capable of detecting any change in the criteria for launch caused by either a direct or nearby lightning strike.

3.7.4.3.3 Spacecraft Charging from All Sources

The system shall operate without performance degradation due to surface charging, bulk charging, and deep charging in accordance with MIL-STD-1541A except paragraph 6.5.2.4.1.

3.7.4.4 Wiring

The power and signal wiring shall be shield twisted pairs with EMI backshells. The shield shall be terminated on the backshell. Each power and signal wire shall have a dedicated return.

3.7.4.5 Grounding

The spacecraft shall have a dedicated external ground. The sensors shall have a single point ground. The spacecraft shall have a single point ground. The impedance between the sensor and spacecraft single point ground shall be less than 10 mega-ohms.

3.7.4.6 Conducted and Radiated Interface Requirements

The interfaces shall meet the requirements (including CE101, CE102, CE106, CS101, CS103, CS104, CS105, CS114, CS116, RE101, RE102, RS101, and RS103) of MIL-STD-461D as tailored by MIL-STD-1541A as tailored within. CS114 and CS116 only apply to power cables. CS103, CS104, CS105, and CE106 apply only to subsystems with transmit/receive antennas. The upper frequency range of RE102 and RS103 shall be extended to envelope all sensor, transmitter, and receiver frequencies.

3.7.4.6.1 Radiated Emission RE101

The requirements shall be tailored by the magnetometer requirements *(TBD)*.

3.7.4.6.2 Radiated Emissions RE102

The requirement shall be less than 100 dBμV/m except as tailored for the search and rescue receiver, the UHF receiver, the SGLS receiver, sensor receivers, and launch vehicle receivers *(TBD)*.

3.7.4.6.3 Radiated Susceptibility RS101

The requirements shall be tailored by the magnetometer requirements *(TBD)*.

3.7.4.6.4 Radiated Susceptibility RS103

The requirements shall be tailored by the external RF environment and the UHF transmitter, the SGLS transmitter, and the launch vehicle transmitter *(TBD)*.

3.8 General Considerations

3.8.1 Spacecraft Reference Coordinate Frame

A right-hand, orthogonal, body-fixed XYZ coordinate system shall be used. The +Z-axis is downward towards nadir, the Y-axis is along the orbit normal (+Y is opposite the orbital angular momentum) and the X-axis is along the spacecraft velocity vector (+X toward the direction of spacecraft travel).

3.8.2 Dimension Unit Standard

All documents shall provide units in metric as the minimum, with English units as an option. All interfaces shall be specified in the international system of units, System Internationale (SI), unless design heritage precludes this. Dimensioning shall be in the as-designed units and identified when other than SI.

3.8.3 Nominal Orbit Parameters

The NPOESS satellite shall operate in a near circular, sun-synchronous orbit. The nominal orbit for the satellite is 833 km altitude, 98.7 (*TBR*) degree inclination. The orbit will be a "precise" orbit (i.e., altitude maintained to ± *TBD* km, nodal crossing times maintained to ± 10 minutes throughout the mission lifetime) to minimize orbital drift (precession). NPOESS must be capable of flying at any equatorial node crossing time. However, the nominal configuration is with the satellite orbits equally spaced, with 0530 and 1330 nodal crossing times for the U.S. Government satellites and 0930 for the METOP satellite.

The sun beta angle, β, is the angle between the solar vector (i.e. the spacecraft-sun line) and the orbit plane. For sensor thermal design purposes, the range of β for the NPOESS missions is ± 90 degrees. The satellite shall maintain the sun on the appropriate side of the satellite to meet the 'all beta' requirement. Sensor design shall allow for approximately a 5 degree infringement of sun on the cold space side of the spacecraft in the case of a noon or midnight orbit.

3.8.4 Sensor/Spacecraft Integration Responsibility

The Government will be the system integrator until a Total System Performance Responsibility (TSPR) contractor is selected in 4Q2000. Until that time the government will be responsible for: accommodation trades, resource allocation (weight, power, space, bandwidth) and resolving interface issues.

3.9 Launch Vehicle Environments

The baseline NPOESS launch vehicle is planned for a medium launch vehicle, most likely a Delta class equivalent. The levels specified in this section reflect the Evolved Expendable Launch Vehicle (EELV) Medium Launch Vehicle (MLV) proposed environments.

3.9.1 Sensor Fairing Dynamic Envelopes

There are standard minimum sizes for payload fairing envelopes (actual fairing envelopes may be larger). These envelopes define the useable volume inside the fairing and forward of the Standard Interface Plane (SIP). However, there will be some stay-out zones in the envelope which currently are *TBD*.

Two sizes of standard fairing envelopes have been defined for MLV, as shown in and Figure 7. Nominal fairing envelope size is as shown in Figure 6. Where the satellite requires more than the nominal size, a larger fairing envelope (similar to Atlas II) shown in Figure 7 is available.

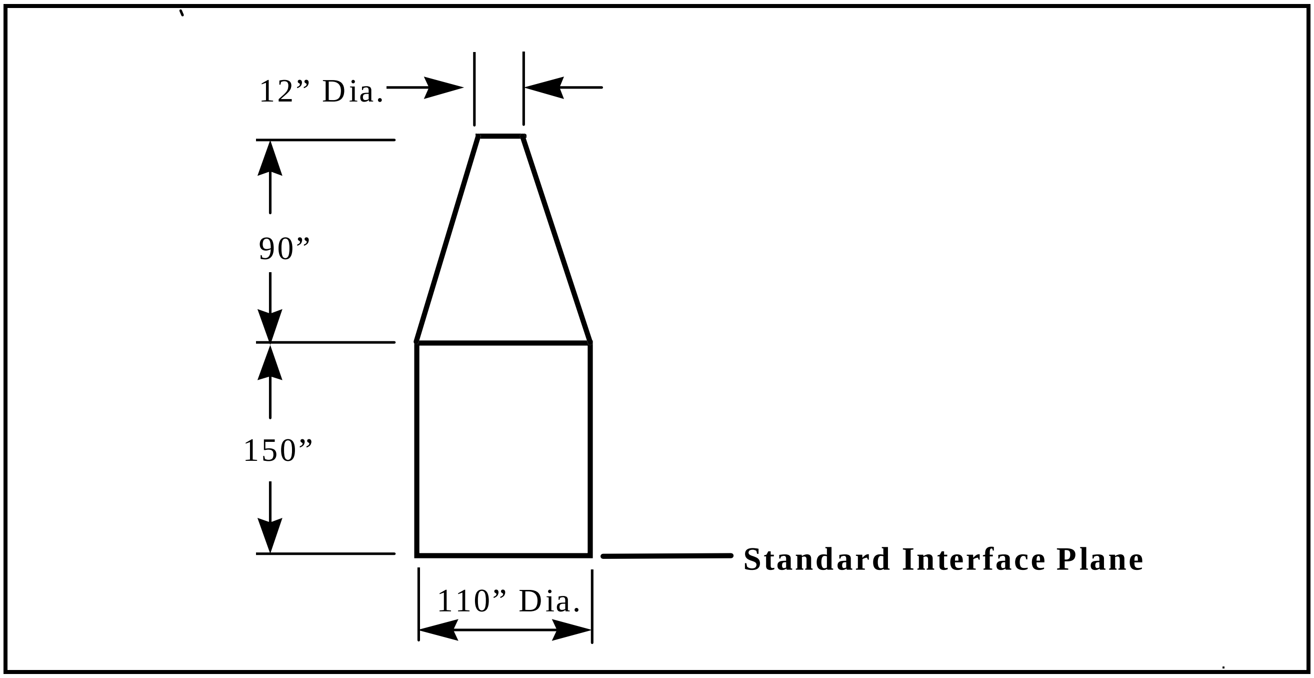


Figure 6. Nominal MLV Payload Fairing Dynamic Envelope

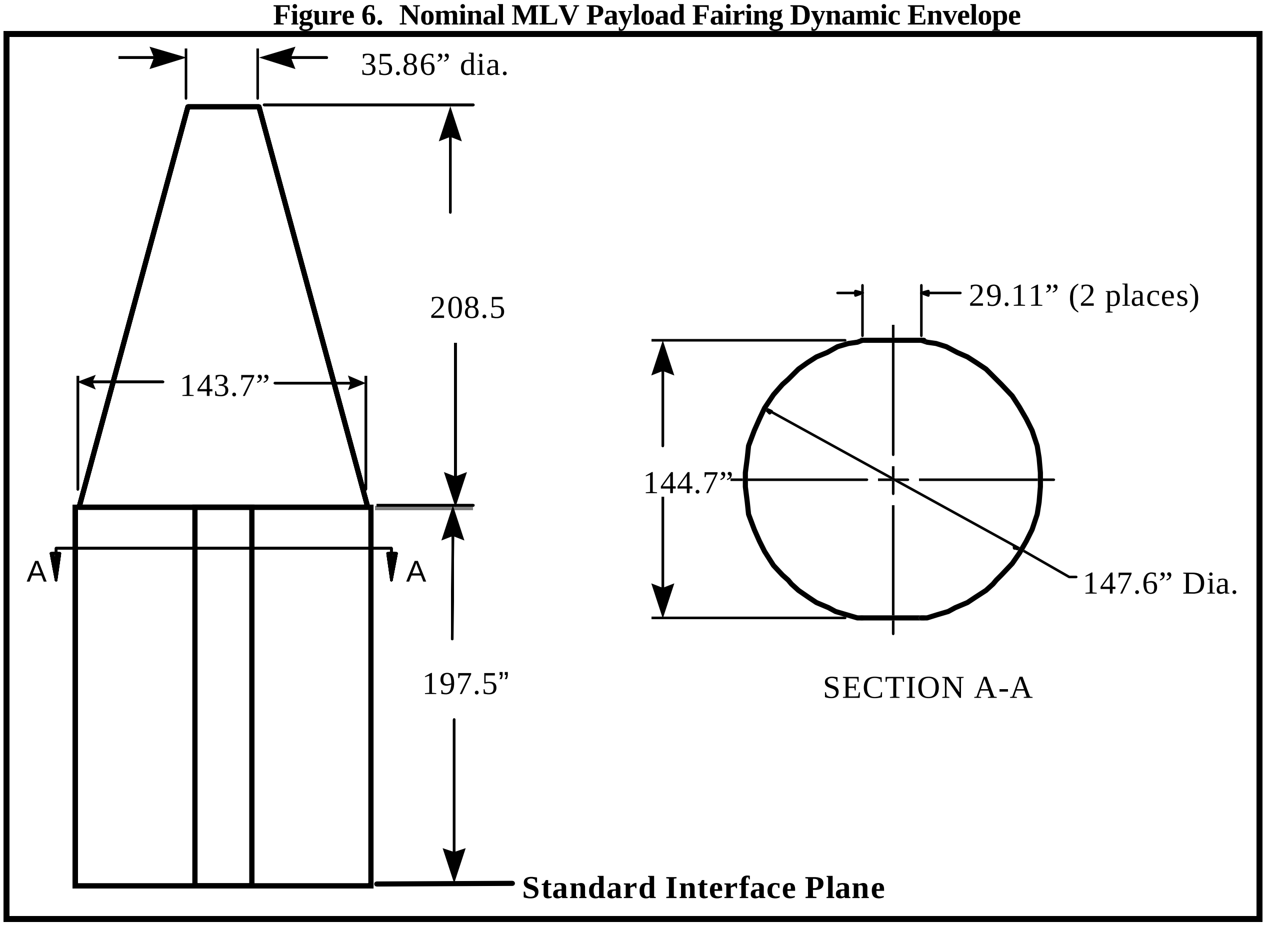


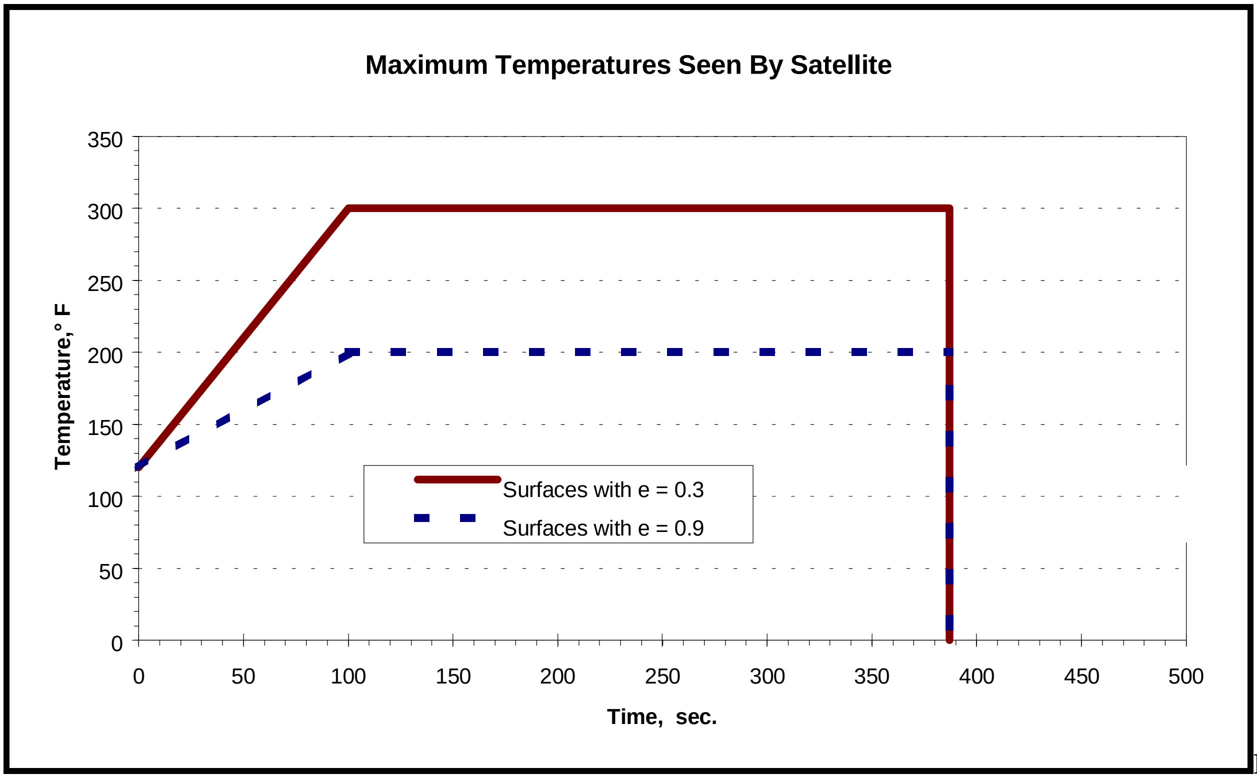
Figure 7 Larger (Optional) MLV Envelope

3.9.2 Thermal

TBS

3.9.3 Temperatures

The worst case effective internal environment in the space vehicle compartment within the fairing during ascent is defined in FiFigure 8. The surfaces seen by the satellite will generally fall into one of two categories: surfaces with low emissivity (e ≤ 0.3) and those of higher emissivity (e ≤ 0.9). Maximum temperatures as a function of the time from launch, 300°F for a surface emissivity of 0.3 and 200° for a surface emissivity of 0.9, are shown in the plot. The exact configuration and percentages of each type of surface is both mission specific and LV specific. Temperatures may exceed those shown but in no case shall the total integrated thermal energy imparted to the spacecraft exceed the maximum total integrated energy indicated by the temperature profile shown in FiFigure 8.



Fi

Figure 8. Maximum PLF Inner Temperatures

3.9.4 Heat Flux

TBS

3.9.5 Free Molecular Heating

The maximum instantaneous 3-sigma Free Molecular Heating on satellite surfaces perpendicular to the velocity vector at the time of fairing separation shall not exceed 0.1814 watts/ cm^2 °C (320 Btu/hr-ft^2)

3.9.6 Shock

The maximum shock spectrum at the SIP (value at 95% probability with 50% confidence; resonant amplification factor, Q=10) shall not exceed the levels shown in Table 4. These levels are shown graphically in Figure 9.

Table 4. EELV Shock Specification

Shock Spectrum from EELV to Satellite (G's)		Shock Spectrum from Satellite to EELV (G's) (due to satellite separation)	
Freq-Hz	MLV	Freq-Hz	MLV
100	40	100	150
125	-	125	175
160	-	160	220
200	-	200	260
250	-	250	320
315	-	315	400
400	-	400	500
500	-	500	725
630	-	630	1100
800	-	800	2000
1600	-	1600	5000
2000	-	2000	5000
5000	3000	5000	5000
10000	3000	10000	5000

Refer to graph of Figure 9 for intermediate frequencies

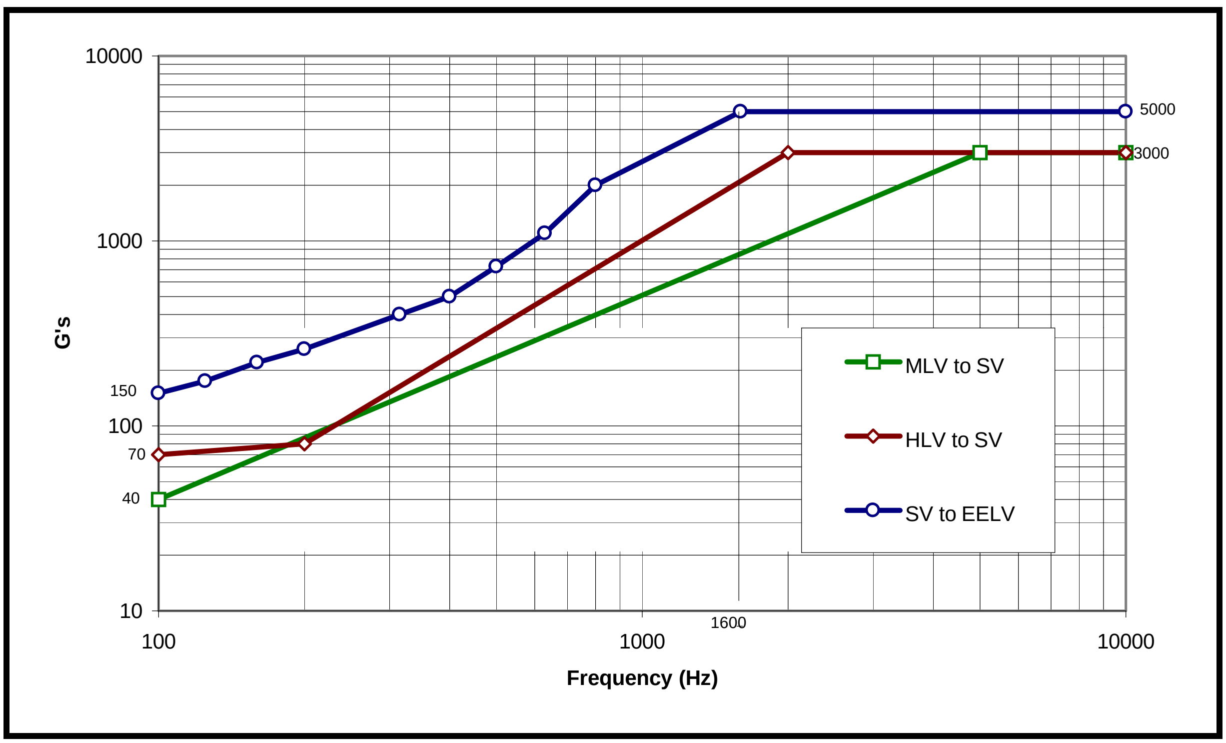


Figure 9 EELV Shock Specification

3.9.7 Launch Pressure Profile

The satellite shall be designed to withstand a payload fairing internal pressure decay rate of 20 mb/sec.

3.9.8 Acceleration Load Factors

Figure 10 define satellite center-of-gravity acceleration values that when used to calculate launch vehicle/space vehicle interface bending moments, axial loads and shear loads will yield values that are guaranteed not to be exceeded. Satellite weights include any payload adapter which may be required. .

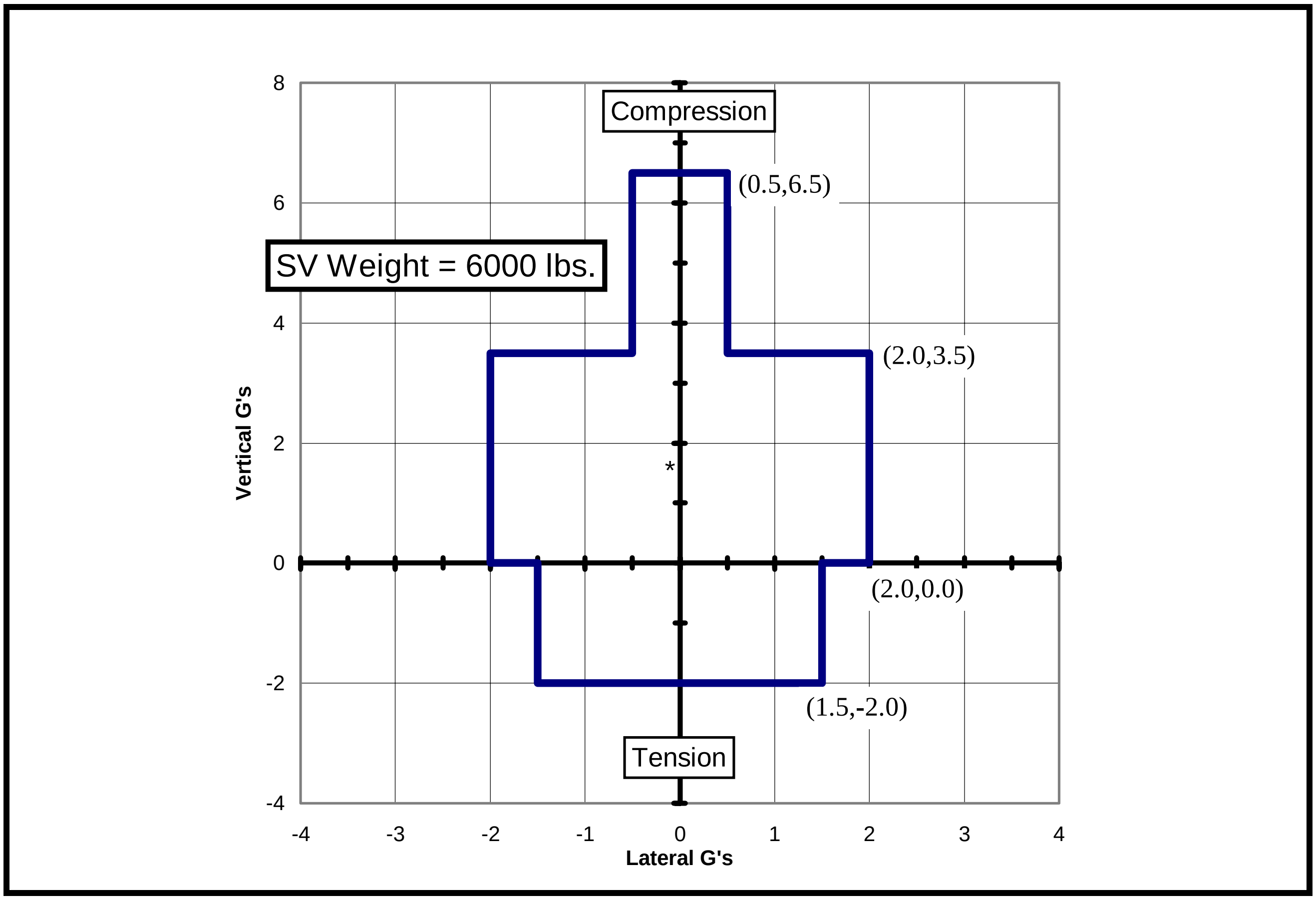


Figure 10. MLV Quasi-Static Load Factors

3.9.9 Vibration

The maximum in-flight vibration levels will be provided in the LV to spacecraft ICD, but are not defined in this document. Space vehicle design should be performed using the EELV acoustic data (provided in the next section).

3.9.10 Acoustics

The free-field maximum predicted sound pressure levels (value at 95th percentile with a 50% confidence), from liftoff through payload deployment shall not exceed those shown in Table 5. These levels are shown graphically in Figure 11 as one-third octave band sound pressure levels versus one-third octave band center frequency for the MLV. The values shown are for a typical satellite with an equivalent cross-section area fill factor of 60%. Higher fill factors may produce higher acoustic levels.

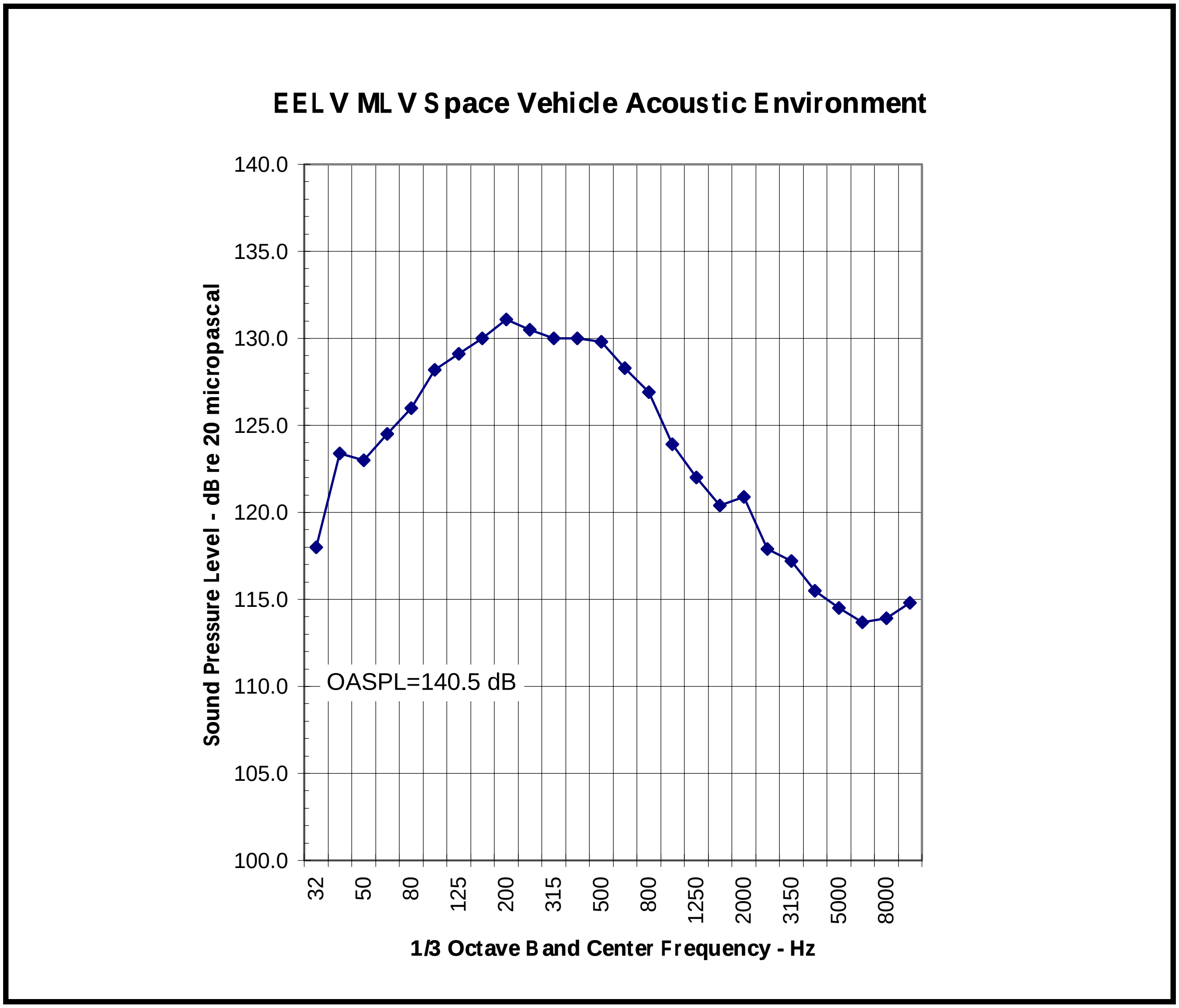


Figure 11. MLV Acoustic Levels

Table 5. Maximum Acoustic Levels

1/3 Octave Band Center Frequency (Hz)	MLV satellite Sound Pressure Level (dB re 20 micropascal)
32	118.0
40	123.4
50	123.0
63	124.5
80	126.0
100	128.2
125	129.1
160	130.0
200	131.1
250	130.5
315	130.0
400	130.0
500	129.8
630	128.3
800	126.9
1000	123.9
1250	122.0
1600	120.4
2000	120.9
2500	117.9
3150	117.2
4000	115.5
5000	114.5
6300	113.7
8000	113.9
10000	114.8
OASPL	140.5

3.10 Math Model Requirements

Thermal and structural models are required for the spacecraft and each of the mission sensors to accurately define the thermal, structural and dynamic loads at each sensor/spacecraft interface.

3.10.1 Finite Element Model

A structural Finite Element Model using NASTRAN is required for each sensor and for the spacecraft. These models are to be used to analyze the structural and dynamic characteristics of each sensor. The models will be used to determine the sensor structural adequacy to withstand transportation, launch and on-orbit loads. Also, they will be used to predict sensor structural resonant frequencies. If the sensor has any structural frequencies less than 50 Hz, a test-verified sensor dynamic (modal) model shall be required. The satellite/launch vehicle loads analysis will be used in the development and definition of the sensor/spacecraft interface loads.

3.10.2 Thermal Math Model

Two mathematical models, provided by the sensor developer, shall be used in defining each sensor/spacecraft thermal interface. A sensor surface geometric math model (GMM) including 50 external surfaces or less, in TRASYS or compatible format, shall be included. A reduced node thermal math model (TMM) of 50 nodes or less in SINDA compatible format shall also be included. (Note: All external surfaces on the sensor shall be represented in the TMM.) Taken together, these two models and the spacecraft thermal model shall fully define the thermal characteristics of the sensor interface in the stowed and (if applicable) fully deployed modes for all operational conditions. The thermal models shall include an adequate level of detail to predict, under worst case hot and cold conditions, all critical temperatures, including those that drive operational and survival temperature limits and heater power. Worst case conditions shall include variations in season, orbit selection, orbital time, environmental flux parameters (seasonal and spatial) and a rational combination of the effects of design tolerances, fabrication uncertainties, material differences and degradation due to aging. Conservative values for conductivity, absorptivity, emissivity, and MLI effective emittance shall be used. Contact resistance shall also be considered. As a goal, to validate sensor-level requirements, all critical nodes predictions shall correlate to within 3^{o}C of thermal balance test data.

The spacecraft and sensor models shall be integrated into a satellite model to define the satellite/launch vehicle interface. All models shall use software and formats that are acceptable to the interfacing contractors and the integrating contractor. All models shall be fully documented to permit ease of use by other contractors in the system.

3.11 Safety Requirements

3.11.1 Design Criteria

All subsystems and interfaces shall be designed to comply with the safety requirements of EWR 127-1.

The use of electro explosive devices (EED's) shall be avoided. Electro explosive devices may be used where use of such devices can be shown to reduce risk. Paraffin and other non-explosive actuators (NEA) shall be activated through the standard command and data interface, and within the sensor envelope. Dedicated EED circuits shall not be included in the baseline standard interface. Space debris shall not be generated.

4.0 TESTING PROVISIONS

A comprehensive sensor test program as defined in MIL-STD-1540C shall be conducted in conjunction with the spacecraft test program to demonstrate that the sensor can meet its performance requirements and ensure that all interface requirements are satisfied. These interface requirements shall include interface structural and thermal loads, electrical power, electrical signals and other interface performance characteristics for ground handling, launch, deployment (where applicable), and on-orbit operations as well as for worst case systems tests conducted after delivery. Many of the tests will be conducted by the sensor developer before delivery of the sensors to the spacecraft contractor. Additional tests will be conducted at the satellite level after integration of the sensor onto the spacecraft. The allocation of tests between the sensor developer and the spacecraft contractor will be coordinated by the integration contractor as part of the interface control function. The coordination of testing shall include such items as sequence of tests, primary test responsibility, test levels, repetition of tests, duration of tests and test location. Acceptance level testing (for workmanship) shall be required on all flight articles except for the protoqual unit. The integration contractor will verify that all required tests are completed successfully. The types of testing to be performed include:

- Thermal vacuum and Thermal cycling
- EMI/EMC characterization to understand and measure radiative and conductive emissions and susceptibility
- Static and Dynamic structural testing (including pressure vessel and ordnance testing)
- Electrical and Mechanical functional testing to demonstrate performance

4.1 Random Vibration

The random vibration test levels are dependent on the payload fairing internal acoustic environment and design of the spacecraft bus. However, the test levels found in Table 6 are considered a conservative estimate of the random vibration environment on a representative spacecraft bus. The test levels shown in Table 6 are the minimum test levels recommended to detect workmanship defects. The test duration shall be 1 minute per axes. In no case shall the test levels for the sensor or its components be less than those shown in Table 6. The protoqualification test duration shall be 2 minutes per axis.

Table 6 Random Vibration - Acceptance Test Levels

Frequency	Acceleration Spectral Density (G^2/Hz)
20	0.01
20 to 160	+3 dB/oct
160 to 250	0.08
250 to 2000	-3 dB/oct
2000	0.01
Overall	7.4 G_{rms}

The plateau acceleration spectral density (ASD) level may be reduced for components between 25 kg and 200 kg according to the component weight (W) up to a maximum of 9 dB as follows:
dB reduction = 10 LOG(W/25)
$ASD_{(plateau)}$level = 0.08 x (25/W)
where W = component weight in kg

The sloped portions of the spectrum shall be maintained at ± 3 dB/oct. Therefore, the lower and upper break points, or frequencies at the ends of the plateau become:

F_L = 160 (25/W) F_L = frequency break point low end of plateau

F_H = 250 (W/25) F_H = frequency break point high end of plateau

The test spectrum shall not go below 0.01 G^2/Hz. For components whose weight is greater than 200 kg, the workmanship test spectrum is 0.01 G^2/Hz from 20 to 2000 Hz with an overall level of 4.4 G_{rms}.

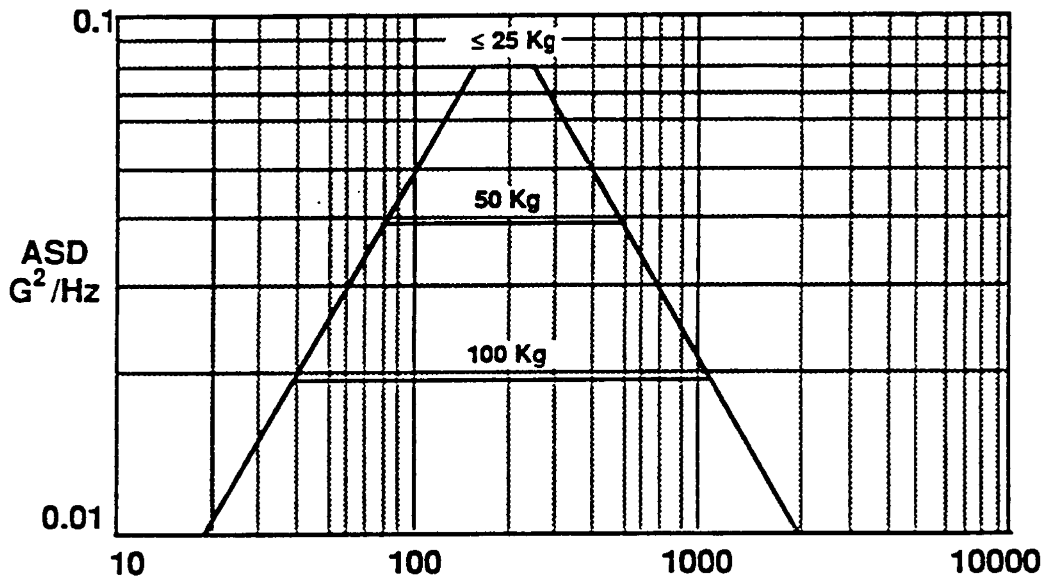


Figure 12 Random Vibration - Acceptance Levels

Table 7 Random Vibration - Protoqualification Levels

Frequency	Acceleration Spectral Density (G^2/Hz)
20	0.026
20 to 50	+6 dB/oct
50 to 800	0.16
800 to 2000	-6 dB/oct
2000	0.026
Overall	14.1 G_{rms}

The acceleration spectral density (ASD) level may be reduced for components more than 25 kg according to:
dB reduction = 10 LOG(W/25)
ASD(25 to 400) = 0.16 x (25/W)
where W = component mass in kg

The slope shall be maintained at ± 6 dB/Oct for components up to 65 kg. Above that mass, the slopes shall be adjusted to maintain an ASD level of 0.01 G2/Hz at 20 and 200 Hz.

For components over 200 kg, the test specification shall be maintained at the level for 200 kg

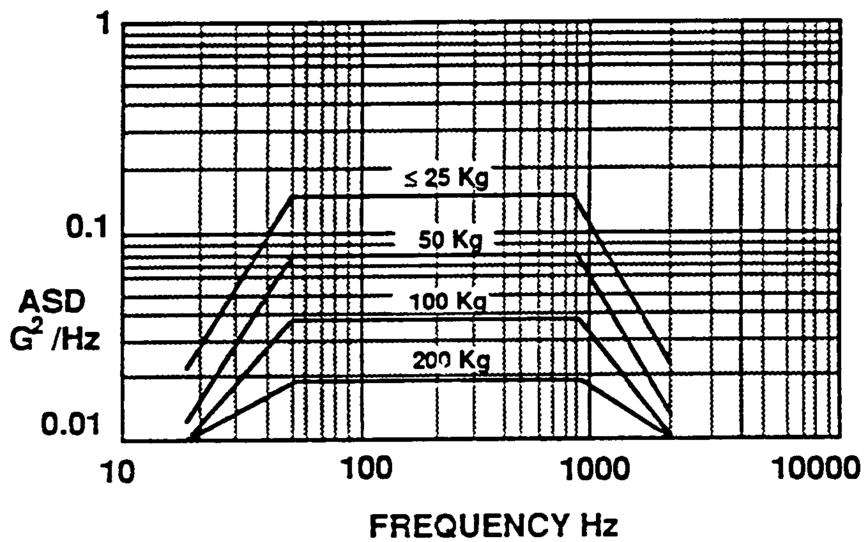


Figure 13 Random Vibration - Protoqualification Levels

4.2 Sine Vibration

The sensor shall be shall be acceptance tested to the sine vibration test levels specified in Table 8 and in Figure 14 in each of three orthogonal axes. During this test the sensor shall be in the launch configuration. There shall be one sweep from 5 Hz to 50 Hz for each axis. The acceptance test sweep rate shall be 4 oct/min except in the frequency range of 25-35 Hz, where the sweep rate shall be 1.5 oct/min. For protoqual testing, the sine vibration levels shall be the same as the acceptance test levels specified in Table 8 however, the sweep rates shall be reduced by a factor of two to 2 oct/min and 0.75 oct/min respectively.

Table 8 Sinusoidal Test Levels

Frequency	Amplitude/Acceleration
5 to 18 Hz	Displacement = 12 mm (double
	amplitude)
18 to 50 Hz	8 G_{peak}

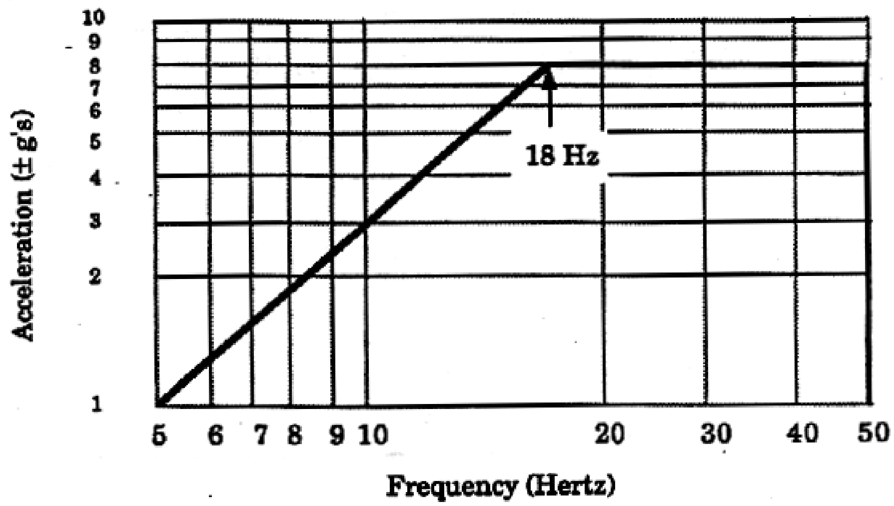


Figure 14 Sinusoidal Protoqualification Test Levels

4.2.1 Design Strength Qualification

The sensor structure shall be tested to a set of loads equal to 1.25 times the predicted loads from a coupled flight loads analysis. These loads may be applied by acceleration testing, static load testing, or vibration testing.

4.3 Acceleration

Sensor flight hardware shall be designed to withstand a maximum acceleration of 0.015g on orbit without permanent degradation of performance.

4.4 Shock

Shock testing is required at the sensor level if there are any self induced shocks (i.e., launch lock releases, pin pullers, etc.). Testing for externally induced shocks (Spacecraft separation, solar array deployment, etc.) is typically accomplished at the spacecraft level. Sensors shall be designed and tested to survive, without permanent performance degradation, the environment shown in Figure 15 to account for externally induced shocks.

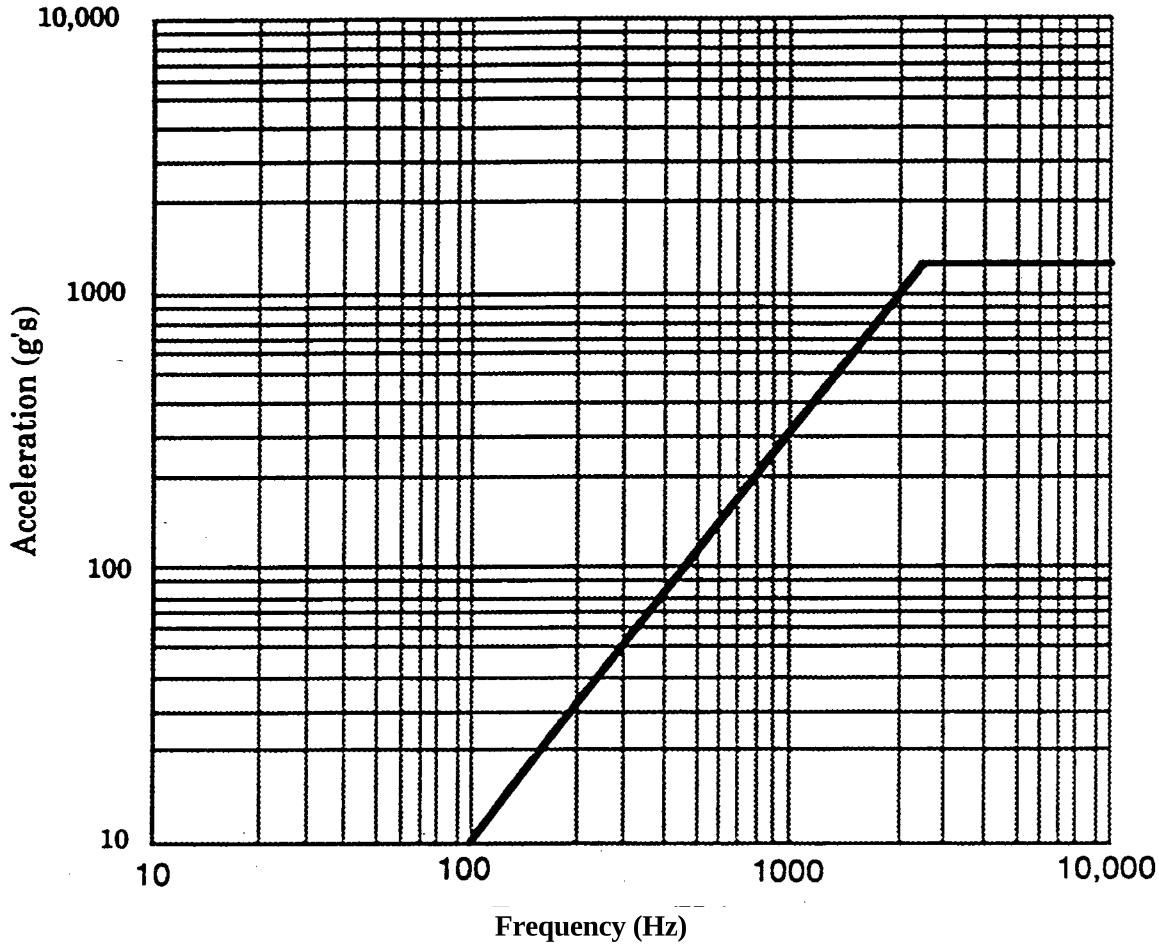


Figure 15 Shock Spectrum (Q=10)

4.5 Protoqualification Level Acoustics

Acoustic testing is required for sensors with large surfaces (units with surface to mass ratio greater than 150 cm^2/kg), which could be excited by the acoustic field directly, and for sensors greater than 180 kg. The acceptance acoustics levels shall be defined in Table 9. The protoqualification levels are increased by 3 dB.

Table 9. Acceptance Acoustics Levels

One-Third Octave Center Frequency (Hz)	Noise Level (dB) ref: 0 dB = 20µPa
25	118
32	123
40	127
50	130
63	132
80	133
100	133.5

125	134
160	134
200	134
250	134.5
315	135.5
400	135.5
500	133
630	128.5
800	127
1000	124
1250	122
1600	120
2000	119
2500	118
3150	116.5
4000	115.5
5000	114.5
6300	114
8000	113
10000	112
Overall	145dB
Acceptance Duration: One Minute Protoqualification Duration: Two Minutes	

4.6 Integrated Spacecraft and Sensor Level Testing

TBD.

4.7 EMC/EMI

Electromagnetic verification testing undergo the following EMI/EMC tests:

1. Conducted susceptibility using CS101 (30 Hz to 100 kHz), CS114 (10 kHz to 400 MHz).
2. Radiated susceptibility using RS103 (20 kHz to 10 MHz).
3. Conducted emissions using CE101 (30 Hz to 20 kHz) and CE102 (20 kHz to 1 MHz).
4. Radiated emissions using RE102 (20 Hz to 10 Mhz).

4.7.1 EMI Testing

Electromagnetic testing shall be performed to verify that the interface will operate properly if subjected to conducted or radiated emissions from maximum expected external sources, and to verify that the design of the interface does not result in deleterious conducted or radiated signals that might affect other mission elements.

Conduction and radiation tests shall be performed on the interface/spacecraft combination and sensor separately, operating with expected power levels, current, and data rates. This segment level testing is independent of the space vehicle-level testing, which also must be performed.

4.7.2 EMC Verification Requirements

All requirements shall be verified by test.

4.7.2.1 System Verification

The spacecraft electromagnetic compatibility margins shall be verified by a system level test in an anechoic chamber. End-to-end RF compatibility shall be demonstrated during this test as well as range and launch vehicle compatibility.

4.7.2.2 Spacecraft Charging Verification

Testing shall be conducted in accordance with MIL-STD-1541A except paragraph 6.5.2.4.1.

5.0 NOTES

5.1 Intended Use

This document is used to establish standard NPOESS spacecraft-to-sensor interfaces and to provide guidance to sensor developers during the risk reduction and design development phases.

5.2 Definition/Glossary

Alignment Knowledge. Alignment knowledge is the angle (arc-sec, zero-to-peak) between the actual orientation of the sensor equipment and the desired orientation.

Alignment Accuracy. Alignment accuracy is the angle (arc-sec, zero-to-peak) between the actual orientation of the sensor equipment and the estimated orientation.

Alignment Stability. Alignment stability is the variation (arc-sec/time interval, peak-to-peak) in the actual orientation of the sensor equipment over specified periods of time.

High Rate Data. Refers to the real time data link to field terminals which contains all channels at the smallest scale horizontal spatial resolution (or cell size) required in Appendix D. Note that the smallest scale horizontal spatial resolution (or cell size) is the same resolution as the “regional resolution” required by the Centrals.

Key Attribute. An EDR attribute that is a key parameter of the system.

Key EDR. An EDR which has a key attribute.

Key Parameter. A parameter so significant that failure to meet the threshold requirement(s) pertaining to its measurement is cause for the System to be reevaluated or the program to be reassessed or terminated. Key parameters include key attributes of key EDRs and the data access requirement. Key parameter requirements are to be included in the Acquisition Program Baseline. (Equivalent to the term “Key Performance Parameter” used in the IORD)

Low Rate Data. Refers to real time data link to field terminals containing fewer channels and/or coarser resolution than the high data rate real time link.

Mission Data. The combination of data provided by any of the mission sensors (i.e. environmental data) plus satellite orbit, attitude, and time tags. It does not include other sensors (i.e. S&R, SDC) or telemetry.

Mission Sensor. Any sensor on the spacecraft directly used to satisfy any of the EDR requirements of the TRD Appendix D.

Orbital Average Power. The value of the power that occurs during normal operation, averaged over one orbit. A calculation to determine this value shall utilize at least 5-minute increments for the duration of the orbit.

Payload. Used to refer to the combination of the mission sensors and the SDC and S&R sensors carried by the spacecraft. Also used to refer to the satellite when it is still mated to the launch vehicle.

Peak Power. The value of the maximum power that occurs during normal operations.

Pointing Accuracy. Pointing accuracy is the angle (arc-sec, zero-to-peak) between the actual orientation of the sensor and the desired orientation of the sensor.

Pointing Knowledge (Real Time or Post-processed). Pointing knowledge is the angle (arc-sec, zero-to-peak) between the actual orientation of the sensor and the estimated orientation of the sensor.

Pointing Stability. Pointing stability is the variation (arc-sec/time interval, peak-to-peak) in the actual orientation of an sensor over specified periods of time.

Protoflight. A protoflight unit is one that was tested to protoqualification levels. The unit is usually the first unit fabricated.

Protoqualification. A test strategy in which qualification and acceptance tests are combined. The protoflight unit is tested to levels beyond what is expected in flight and minimum workmanship levels, with test levels and duration less than qualification levels and duration.

Satellite. The spacecraft and its sensor payloads.

Sensor. The mission-peculiar equipment or instrument to be manifested on a given space mission. The requirements specified apply to individual sensor interfaces, not the total sensor complement.

Sensor Suite. One or more sensors needed to satisfy the EDR requirements allocated to a given Sensor Requirements Document (SRD). It does not include sensors from other SRD suites which provide secondary data contributions to those EDRs.

Spacecraft. The components and subsystems which support the sensor(s) and provide housekeeping functions such as orbit and attitude maintenance, navigation, power, command, telemetry and data handling, structure, rigidity, alignment, heater power, temperature measurements, etc.

Standard Interface Plane (SIP). The SIP is the plane which defines the interface between the LV-provided and the spacecraft-provided equipment.

Tailoring. Modification of this guideline to maximize utility, considering design complexity, mission criticality, cost, and acceptable risk.

TBD. Applied to a missing requirement means that the spacecraft and/or sensor contractor should determine the missing requirement in coordination with the government.

TBR. The requirement will be resolved (*TBR*) between the contractors and government.

TBS. The government will clarify or supply the missing information in the course of thie contract.

Transients. Short-duration changes in the power drawn by a component. Transients are aperiodic and include non-recurring current surges and voltage spikes.

5.3 Acronyms

AWG	American Wire Gauge
BC	bus controller
BTU	British thermal unit
CCSDS	Consultative Committee for Space Data Systems
c.g.	center of gravity
CVCM	collected volatile condensable material
dB	decibel
dc	direct current
DoC	Department of Commerce
DoD	Department of Defense
EED	electro explosive device
EMC	electromagnetic compatibility
EMI	electromagnetic interference
ESD	electrostatic discharge
FEM	finite element model
FMH	free molecular heating
FOV	field of view
GMM	geometric math model
GN&C	guidance, navigation, and control
GPS	Global Positioning System
Hz	hertz
ICD	interface control document
IDD	Interface Design Description document
I/F	interface
IR	infrared
k	kilo (1000)
kbps	kilobit per second
kΩ	kilo ohms
LV	launch vehicle
mb	millibar
M	mega (1 million)
MMA	moving mechanical assembly
MLI	multilayer insulation
MLV	medium launch vehicle
MΩ	mega ohms
NASA	National Aeronautics and Space Administration
NASTRAN	NASA structural analysis
NEA	nonexplosive actuator
OASPL	overall sound pressure level
P/L	payload
RINEX	receiver-independent data exchange

rms	root-mean-square
RT	remote terminal
S/C	spacecraft
SINDA	systems improved numerical differencing analyzer
SIP	standard interface plane
SOH	state of health
TBD	to be determined
TBR	to be resolved
TML	total mass loss
TMM	thermal math model
TRASYS	thermal radiation analysis system
UTC	universal time code
v	volts
w	watts